
Chapter 7

Materials for Launch Vehicle Structures

*Grant Henson
Chief Scientist
Invariant Laboratories LLC
Westlake, Ohio*

7.1 Introduction

This chapter concerns materials for expendable and reusable launch vehicle (LV) structures. An emphasis is placed on applications and design requirements, and how these requirements are met by the optimum choice of materials. Structural analysis and qualification strategies, which cannot be separated from the materials selection process, are described.

A launch vehicle is an airborne system that delivers a payload from the ground to suborbital, orbital or interplanetary space. The payload is usually housed in a space vehicle or satellite that is not considered part of the LV. When it is not important to distinguish the payload from the space vehicle, both may be referred to as the payload.

Modern LVs are designed with a particular type of payload in mind (astronauts, earth-orbiting instruments, interplanetary probes, etc.) but

7. Materials for Launch Vehicle Structures

at the dawn of the Space Age, vehicles performed multiple duty. For example, the Atlas, Titan, and Thor/Delta vehicles all began as long-range weapons and were later adapted for orbital delivery. Sounding rockets such as Aerobee (historical) and Black Brant can leave the atmosphere but do not enter orbit. For the purpose of this chapter, shorter-range missiles that never leave the atmosphere are not considered LVs.

Most LVs, including Atlas, Delta, Ariane and Proton are expendable. Expendable vehicles are flown only once; the upper stages may be disposed of through a controlled re-entry, or may be left in orbit as “space junk,” whereas the first stage or booster falls to earth in a cleared area. The term booster usually means the first stage of a multi-stage LV and will be used in that sense here.

Reusable systems may incorporate a single vehicle that both launches the payload and houses it while in space, the prime example being the Space Shuttle Orbiter. The Orbiter, and the similar Soviet Buran vehicle, are here considered LVs rather than space vehicles, because they must sustain atmospheric flight loads and environments similar to those sustained by expendable boosters. Therefore, the materials selection aspects are much the same as for expendable LVs. Proponents of reusable vehicles assert that they can be cheaper and more reliable than expendables. On the other hand, recovery and refurbishment are costly, and a failure of a vehicle intended for re-use is more damaging to schedules and budgets than a failure of an expendable vehicle. The envisioned benefits of reusability have led to recent investment, both public and private, in reusable vehicle development.

One source [1] claims that a reusable variant of the Aerobee sounding rocket was flown; if so, it was the first reusable vehicle. Notable reusable orbital LV programs that never demonstrated powered flight were the Sea Dragon, X-33, X-34 and the K-1. The first stage of the Soviet/Russian Energia vehicle, developed to lift the Buran orbiter as well as other heavy orbital payloads, was designed to be reusable for at least ten flights [2]. However, it has never actually been recovered and reused. The DC-X/-XA was an early demonstration of reusable rocket flight within the atmosphere. SpaceShipOne reached suborbital space in 2004, landed, and repeated the feat. However, neither of these systems led to a sustained record of operations. In 2015, a New Shepard vehicle, including both the booster stage and the space vehicle, was recovered from suborbital flight

and then successfully reflown 61 days later. Also in 2015, the booster stage of a Falcon 9 was recovered by powered descent onto a land-based pad after having launched a payload to orbit; since then several attempts to descend onto a seagoing platform have been successful.

Today, space launch vehicles are considered, along with aircraft, part of a single endeavor we call “aerospace.” But various dictionaries date this term only back to the late 1950s, at least a decade after the guided missile, for better or worse the archetype of the modern LV, was developed. In most nations, the initial authority for developing guided missiles rested with the artillery or ordnance corps, not the air corps. The relevance of this observation is that while launch vehicle materials and structures technologies have much in common with those of aircraft, the degree of commonality is perhaps less than one might think.

Investment in LV development and operation is now a small part of the overall aerospace economy.¹ However, for several decades, political and military imperatives drove high expenditures on LV development, leading to significant advances. New materials and structures had to be developed in parallel with other vehicle systems in “crash” programs, under high risk of technological failure, in order to satisfy aggressive performance requirements within the desired time frame. While the pace of innovation was slow for decades, increased emphasis on cost reduction and improved reliability continue to drive incremental advances in materials and structures technology. Also, large, qualitative improvements in computing capabilities and newly available precursor materials have provided a technology push to encourage further advances in LV materials and structures.

Because materials selection for LVs is affected by laws and regulations that vary from country to country, it is important to note where LVs are built and used. Until the 1970s, the United States and the Soviet Union (Russia and Ukraine) dominated LV production. More recently, France

¹For example, the U.S. Census Bureau reported about \$23 billion in deliveries of “guided missile and space vehicle manufacturing,” “guided missile and space vehicle propulsion unit and propulsion unit parts manufacturing,” and “other guided missile and space vehicle parts and auxiliary equipment manufacturing” in 2005, which surely includes many billions spent on non-launch-vehicle hardware such as anti-aircraft missiles. Compare this to \$114 billion in deliveries of aircraft and related items [3]. Considering that many countries manufacture aircraft but not launch vehicles, LVs probably constitute under 10% of the global aerospace economy.

7. Materials for Launch Vehicle Structures

and China have developed and operated a significant number of LVs. Within the last few years, India, South Korea, North Korea and Iran have also developed LVs. The French and Ukrainian vehicles are launched from different countries than the ones they are produced in. Also, many vehicles contain major substructures or engines built in several different countries. Table 7.1 shows orbital launches broken down by country of final factory assembly.

Table 7.1: Orbital vehicles launched over two recent periods, grouped by country of production [4, 5].

Period	Country of Production	Share
1990 to 1998	US	39%
	Russia	32%
	France	13%
	Ukraine	9%
	China	5%
	Japan	2%
	Israel, India	< 1%
2007 to mid-2009	Russia	30%
	US	26%
	China	14%
	Ukraine	13%
	France	8%
	India	4%
	Japan	2%
	Iran, Israel, North Korea, South Korea	< 1%

Chapters 11 and 12 of this book are dedicated to materials for the solid rocket motors and liquid rocket engines, respectively, that propel LVs. Propulsion materials and structures are mainly affected by the loads and environments generated within the engine or motor itself, such as thrust chamber pressure. However, a section is provided in this chapter on large solid rocket motor cases, because they can form a significant part of the load-bearing capability of the vehicle as a whole. The structural failure of a large strap-on solid rocket motor on an Ariane 5 or the Space Shuttle, or the solid rocket boost stage of the Ares I, would doom the vehicle structure rather than just the propulsion system. Inclusion of solid rocket motor cases with the structural system

rather than the propulsion system follows the precedent set in [6]. Also, propulsion support systems such as propellant feedlines are included in this chapter, because they are usually the responsibility of the launch vehicle contractor.

7.2 Launch Vehicle Structures

The typical missile-derived expendable LV may be thought of as a stack of tanks with an engine at one end and a payload at the other. The fuel and oxidizer are contained in separate tanks. In more detail, the engines are mounted to the aft end of the tanks and exert thrust through a reinforced structure. The tanks are connected with thin-walled cylinders called *skirts* or *intertanks*. Complete stages are connected to one another through cylindrical shells called *interstages* or *adapters*. When the connected stages are of different diameters, the adapter has the shape of a truncated cone, which may have its smaller diameter forward or aft. When the smaller diameter is aft, the structure may be referred to as a *boattail*.

The forward end of the vehicle is formed by a tapered shell that also encloses the payload. This structure is referred to as the *payload fairing*, *payload shroud*, *nose fairing*, or *nose cone*. Inside the nose cone, and attached to the forward end of the upper stage, is the payload. The payload is attached through a *payload adapter* or *payload fitting*. Therefore, at the forward end of the vehicle, there are two primary load paths: the payload fairing or *outer branch* and the payload attach fitting or *inner branch*. Usually, but not always, the tank walls themselves carry the primary loads. Occasionally, if a stage is much smaller in diameter than the payload compartment or the booster, the entire stage may be contained in a non-load-bearing *aeroshell* or *aerofairing*.

The major substructures are attached using bolted flanges. The connections may be made with the vehicle in either the horizontal or vertical position, in a factory or at the launch site. The final placement of the payload onto the vehicle frequently takes place with the vehicle actually sitting on the launch pad.

Figure 7.1, a cutaway view of the Saturn V launch vehicle used to launch astronauts to the Moon, shows the location of the tanks, engines, and payload. The Apollo payload was unusually large and bulky, and resided within a complex fairing topped by an escape rocket. A nearly

7. Materials for Launch Vehicle Structures

cylindrical interstage can be seen joining the booster to the second stage, and a conical one can be seen joining the second and third stages. Some internal structures and stiffeners in the tanks are visible. The booster fuel and oxidizer tanks are joined by a cylindrical intertank, while the second and third stage tanks have common bulkheads to save weight and volume.

The *outer mold line* is the outermost surface of the cylindrical structure, visible from the outside, while the *inner mold line* is the inner surface. These terms, common in composite molding processes, are used even if there was actually no molding involved in building the structure.

LV shell structures may completely lack internal bracing or stiffening, may have stiffeners integrally machined into the wall, or may have mechanically attached stiffeners or braces. Extensive internal framing is rarely used in launch vehicles except in thrust structures.

The term *membrane* is used to refer to the part of a shell structure far from attachments or other discontinuities, in which only in-plane loading is significant. This same area may be called *acreage*, especially when discussing thermal protection systems. In contrast, flanges, door seals, bolt lines, and the like may be called *details* or *closeouts*; closeouts especially refer to small items or fasteners that are the last to be installed when building the vehicle.

Reusable designs with winged launch and re-entry vehicles do not conform to the description just given. The Space Shuttle is functionally split into the reusable Orbiter, the partially reusable Solid Rocket Boosters, and the expendable External Tank (ET). Many different concepts, from single-stage-to-orbit to staged systems comprising a winged vehicle piggy-backed on a more conventional missile-like booster, have been proposed. Wilhite [7], in the context of a particular trade study, discusses some of the materials selection aspects of advanced fully reusable designs. It is telling that only rather exotic materials (a metal matrix composite with silicon carbide fibers, and monolithic titanium aluminide) were considered feasible for the two-stage-to-orbit systems he explored.

7.3 Basic Material Characteristics

As with other aerospace applications, the most important characteristics of LV materials are

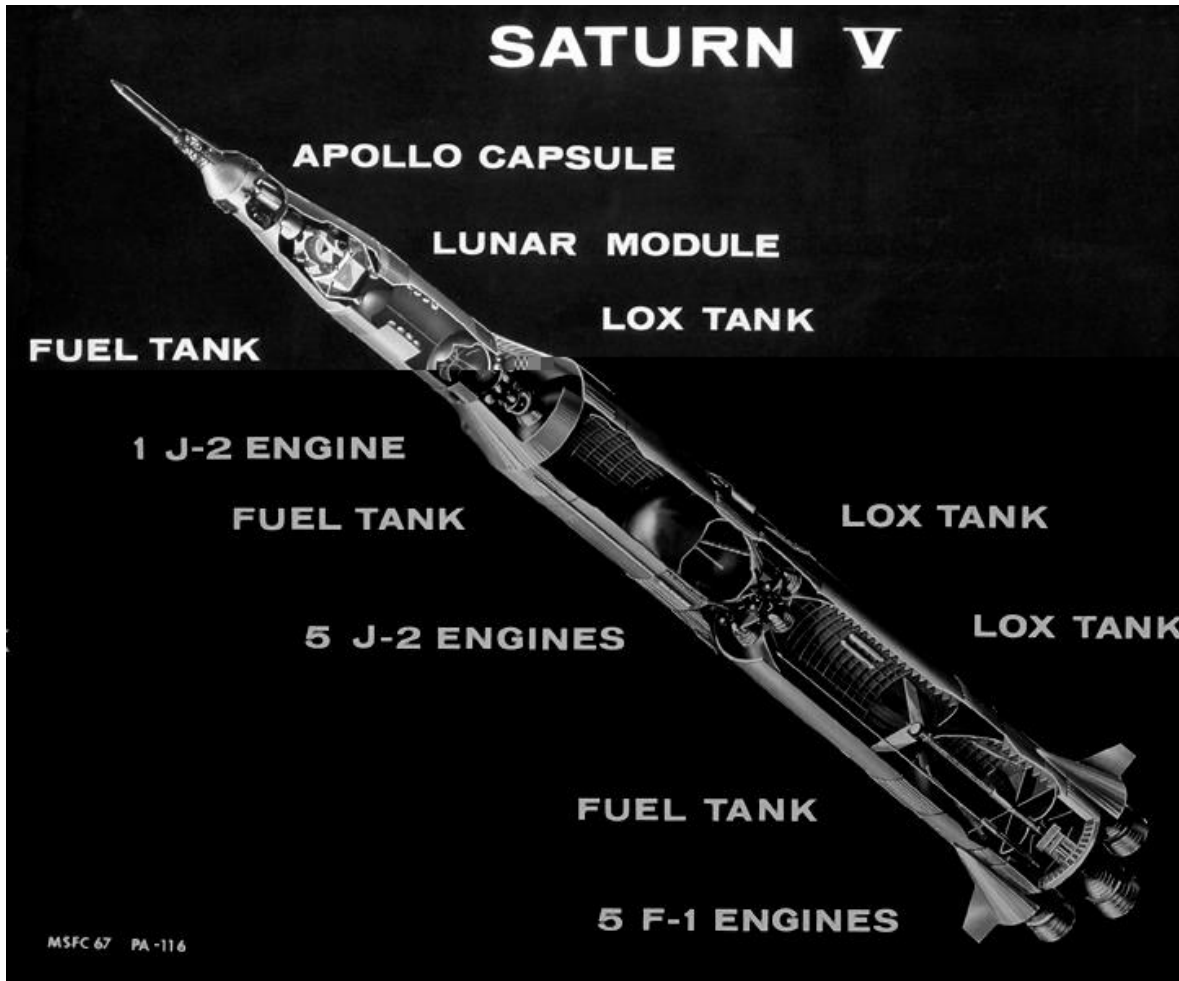


Figure 7.1: Cutaway view of the Saturn V launch vehicle with the Apollo payload, showing major substructures. NASA graphic.

- material strength, based on any applicable failure criteria,
- material stiffness, as quantified by the elastic modulus or moduli,
- mass density,
- nature of the failure modes (gradual or sudden),
- ability to tolerate small-scale damage,
- mechanical and chemical compatibility with nearby materials.

Long-term damage resistance or durability are not as important in ex-

7. Materials for Launch Vehicle Structures

pendable LVs as in reusable ones, and much less important than in aircraft.

In many LV applications, the foregoing must remain favorable at very high or low temperatures and in the presence of humid, corrosive or other degrading environments. Because most launch vehicles use cryogenic propellants, properties at very low temperatures are important; high-temperature properties can also be important because of the aerodynamic heating encountered in the high-speed atmospheric part of the trajectory.

Knowledge of material characteristics must be quantitative in order to play a direct role in structural system trade studies. The stiffness and density of most materials are consistent enough to be treated as deterministic values for a particular material at a given temperature. However, material strength displays sample-to-sample variation that must be taken into account in both design and analysis; design values based on tenth- or first-percentile strength are more important than average strength. Further, if the factors tending to cause variations in strength are poorly understood, high safety factors must be used to preserve reliability, leading to heavier structures.

Equally important is manufacturability. Without the ability to shape or assemble a material into an efficient structure, the material's intrinsic advantages become meaningless. For instance, a single carbon nanotube is extremely strong, but until a carbon nanotube structure of useful size can be manufactured while preserving this extreme strength, that material will not play a significant economic role. Aspects of manufacturability that are especially relevant to LV applications include

- weldability,
- machinability,
- ease of making a composite laminate, and formability or “drape” of plies,
- ease of assembly using fasteners, co-curing, adhesives, locking features and so on.

Thermal properties may also be important; in particular, it is desirable to have thermal expansion characteristics that are predictable and compatible with adjacent materials, including tooling.

These general characteristics must be associated with relevant, measureable material properties, or at least be translated into standardized tests. A good summary of the properties and tests most relevant to structural design can be found by reviewing the data tables in the universally referenced *Metallic Materials Properties Development and Standardization* (MMPDS) published by the Federal

Aviation Administration (FAA) [8]. This reference was formerly known as MIL-HDBK-5. In this work we find data on

- material strength, including typical values and statistically derived lower-bound design allowables for
 - tensile yield and rupture (“ultimate”)
 - compressive yield
 - shear rupture
 - bearing yield and rupture
- elongation to break
- tensile and compressive Young’s modulus
- shear modulus
- Poisson’s ratio
- density
- thermal conductivity, heat capacity, and thermal expansion coefficient

These properties are reported for a wide range of tempers of commonly used metals. They are usually given for various thicknesses, because heat or age treatment affects metals differently depending on the thickness. Also, they may be given at elevated or cryogenic temperatures for various exposure times, or plots of temperature adjustment factors may be provided. In some cases, full-range stress-strain curves are provided. These are required in order to perform stress analysis in the plastic range. Finally, S - N (fatigue) diagrams and Paris-region crack growth curves are provided for many alloys.

Metal properties at cryogenic temperatures depend strongly on the crystal structure. Face-centered cubic metals such as aluminum and the austenitic stainless steels experience a rise in ultimate strength but a lesser increase in yield strength, which preserves their ductility. Body-centered cubic metals such as the ferritic steels tend to experience a greater increase in yield strength than in ultimate strength, which results in more brittle behavior.

For composite materials, which are generally not isotropic, more extensive (and expensive) testing may be required for full characterization. To take full advantage of the directional stiffness and strength properties of composites, directional material properties must be available. Composite

7. Materials for Launch Vehicle Structures

properties are not as readily available as metal properties, because of the proprietary constituents and processes that are used, and hence are not widely applicable. However, one frequently consulted reference that may be used for initial design calculations is the *Composite Materials Handbook* [9], formerly sponsored by the Department of Defense as MIL-HDBK-17. In this handbook we find data on strength, modulus and elongation to break for fiber, tapes, prepreg cloth and laminae, under various temperature and moisture conditions. This information, in combination with thickness and ply angles for laminate designs, may be used to build up the full laminate stiffness matrix. Much of these data are labeled by fiber volume fraction, ply thickness, and other processing parameters, but these parameters may vary so much in practice that it may be difficult to find directly applicable handbook data.

MMPDS defines the A-, B- and S-values as statistical minimums for design use. Roughly speaking, the A- or S-values are suitable for non-redundant structure and the B-values are suitable for redundant structure. The A-value is the value that 99% of all samples are expected to exceed, at the 95% confidence level. The B-value is the value that 90% of all samples are expected to exceed, at the 95% confidence level. The S-value is not a statistically derived value but rather a specification minimum. S-values may be substituted for A-values provided the material is screened to ensure the S-value is met.

While every materials and structures engineer should be thoroughly familiar with these definitions, their significance should not be exaggerated. It has been said that “typically, less than 1 percent of composite structures on large aircraft is actually governed by unnotched laminate strengths” [10]. While this may be overstating the case, it is clear that the familiar uniaxial tensile strengths are not the last word in material characteristics. Reference [10] states that “joints, damage tolerance, and stiffness” govern the choice of the rest of the materials.

The above may be regarded as a minimum set of properties needed to produce a credible preliminary design. However, many other properties, in particular strength properties under flight-like combinations of loads and including stress raisers, are important. Even with the widespread availability of finite element analysis, it is still important to characterize material strength in realistic regimes through careful testing. A detailed, nonlinear, validated finite element analysis may well prove more expensive

and less reliable than a well-planned test to determine, for example, the fatigue life of a bonded joint. Some examples of strength testing from the literature are biaxial strength [11], cryogenic fracture toughness and fracture toughness ratio [12], hardness, tangent modulus, impact, notched fatigue, weld coupons, and creep-rupture [13].

In addition to numerical property data, MMPDS and the *Composite Materials Handbook* also include information on applications, material processing, corrosion resistance, maximum service temperatures, and other information relevant to the designer.

A comprehensive handbook on materials selection for launch vehicles (and space systems in general) that is more oriented toward physical/chemical properties and compatibility is MSFC-HDBK-527, *Materials Selection List for Space Hardware Systems*, published by NASA Marshall Space Flight Center [14]. This handbook provides a very extensive summary of knowledge concerning the corrosion, stress corrosion cracking, propellant and working fluid compatibility, flammability, toxicity and thermal vacuum stability properties of aerospace materials, both metallic and nonmetallic.

Another excellent reference is the *Aerospace Structural Metals Handbook* [15]. This work, which was originally sponsored by the Air Force Materials Laboratory, contains not only extensive tables of data, but also a cross-reference so that the same alloy may be located under names that may vary from producer to producer or country to country. Data are usually typical properties rather than statistical minimum design values. The book is now available as an online database.

Per-piece raw material cost is usually small compared to tooling and labor costs at the low production rates typical of LVs. Therefore, the cost of the material *in its unprocessed form* is rarely an important consideration in materials selection. If a material is commercially available in the required sizes, quantities, and on the needed schedule, it is a candidate for use in a launch vehicle structure, practically regardless of cost. Historically, space programs would even specify custom materials having no existing commercial applications and therefore being subject to unknown cost and production fluctuations; for example, Rocketdyne developed NARloy-Z specifically for use in the linear aerospike engine and Space Shuttle Main Engine [16]. But lately this high-risk, high-reward approach has been discouraged.

7. Materials for Launch Vehicle Structures

Although much of the effort to develop requirements and materials for reusable vehicles stemmed from the Space Shuttle Orbiter program, the same questions had to be addressed by the designers of the Soviet/Russian Buran orbiter [17]. Much of this development had to take place independently, because of the political situation. Unlike the Space Shuttle Orbiter, Buran did not have booster engines, only orbital maneuvering engines; launch was solely by means of external boosters. The Buran designers found that riveting was not compatible with graphite-epoxy composites, due to inadequate impact strength. They also reported that due to galvanic corrosion, it was not possible to use aluminum fittings with composites, so titanium was used instead. This problem was largely solved on the Space Shuttle by careful material compatibility studies. As in the West, the Buran designers noted that the strength and stiffness properties of composites tend to vary more than those of metals. Finally, the Buran designers identified fastening and joining as the key challenge in designing with composites, a finding that many composites designers will agree with.

Durability and Reusability

Fatigue, fracture and aging characteristics are less important for expendable launch vehicles than for aircraft or reusable LVs. However, when long delays between manufacture, testing and operation must be accommodated, thermal and chemical aging as well as ambient moisture uptake should be considered in materials selection. Repeated ground tests can consume some of the fatigue life. Material characteristics that are particularly important in reusable vehicles are

- resistance to fracture and the propagation of cracks under fluctuating loads
- ductility
- resistance to stress corrosion
- the ease with which damage can be found and characterized, and
- chemical and electrochemical compatibility with other materials or contained fluids.

Structures in a reusable vehicle will obviously experience more loading cycles than if the vehicle were expended, but airliner-style operations

in which thousands of flights may be accumulated are not yet possible for LVs. For example, the Space Shuttle Orbiter airframes had a design lifetime of 100 missions. For metal primary structure not exposed to high load fluctuations, and designed to withstand flight loads without macroscopic yielding, 100 missions will not consume a significant amount of the high-cycle fatigue life. However, undetectable pre-existing cracks on highly loaded structures or near stress raisers may grow to dangerous lengths within 100 flights. Failures due to fracture may pose a risk to nearby components if a moving part is liberated. Also, low-cycle fatigue, which by definition requires significant plastic deformation, can be important on expendable LVs.

In the present context, it is sufficient to understand that the fracture failure mode occurs when a fatigue crack grows to its critical size (the size at which unstable, catastrophic propagation of the crack occurs). Predicting the initiation of a crack is outside the normal scope of the fracture analysis; the analysis assumes the existence of the largest undetectable crack at the worst-case location at the time of inspection. The fracture or “safe-life” analysis predicts the growth of the crack under the expected “spectrum” of fluctuating loads. It predicts how long the loads may be sustained before the crack reaches its critical length.

Safe-life analysis² may be defined as the understanding and quantification of life estimates. Safe-life-critical structures are likely to be included in the LOLI (Limited Operating Life Item) listing of the vehicle. LOLI hardware can be life-limited due to corrosion life, battery life, time of operation, thermal cycles, etc., but here we focus on the safe-life fracture analysis. A LOLI definition is provided for the vehicle which includes the “zero time,” and how cycles are to be counted. A quality control group tracks the cycles for each vehicle. As far as safe-life is concerned, LOLI counts are counts of stress excursions beyond a defined level, and the zero time is the time at which flaw inspection was done. Re-inspecting the structure is a way to reset the zero time and gain additional life.

For an expendable vehicle, the service life is an assumed, fixed number of load cycles high enough to allow checkout and multiple launch attempts, each involving a load cycle due to tank prelaunch pressurization. As more and more vehicles of a particular type are launched, fewer launch

²This material on fracture-based safe life and fracture control was contributed by John Hilgendorf, Structural Analysis Lead for Delta II, United Launch Alliance.

7. Materials for Launch Vehicle Structures

attempts should be needed per actual launch, so the assumed service life may decrease. For life-limited structures, the shorter assumed service life can lead to higher life margins, greater tolerance for manufacturing discrepancies or found flaws, and lighter-weight structure in case there is an opportunity for design changes.

Figure 7.2 shows an idealized crack growth curve for a metal under fluctuating stresses. This is commonly referred to as a da/dN curve, where a is the crack length and N is the number of cycles. The many factors influencing this curve, such as stress ratio and frequency, are discussed in detail in previous chapters. Due to the short life of an expendable LV, crack-growth concerns are frequently in Region 3 of the da/dN curve.

Being unstable in nature, Region 3 predictions can be unreliable. When the metal is ductile, much of this Region 3 crack growth is of a tearing nature. In situations where production discrepancies or damage during pre-launch operations occur, it is sometimes necessary to remove conservatism to adequately assess the risk associated with the damage. In these cases, elastic-plastic fracture mechanics or other less conservative theories may be used.

When sustained loading is part of the load spectrum, stress corrosion of the potential flaw needs to be considered. K_{eac} (or K_{Issc}) is a truncated value which toughness can be degraded to, under sustained loads. The stress corrosion resistance may need to be taken into account for pressure vessels storing fluids used to pressurize pneumatic, hydraulic or ullage pressure systems. The time at load can be as short as a few hours.

For vehicles considered to be at risk of failure due to crack propagation, a formal fracture control program may be implemented. Information describing how to write a fracture control plan may be found in [18]. A fracture control program classifies parts as fracture-critical if they exceed a certain mass, are uncontained, non-fail-safe, part of a pressurized system, or meet other criteria that suggest serious consequences in case of failure. For fracture-critical components, the fracture control program applies special analysis, testing and inspection requirements to reduce the chance of a harmful fracture. These vary from program to program but generally amount to an analytical determination of the smallest crack that could grow to critical size before the next regular inspection, and an inspection plan that will detect a large percentage of cracks larger than that critical

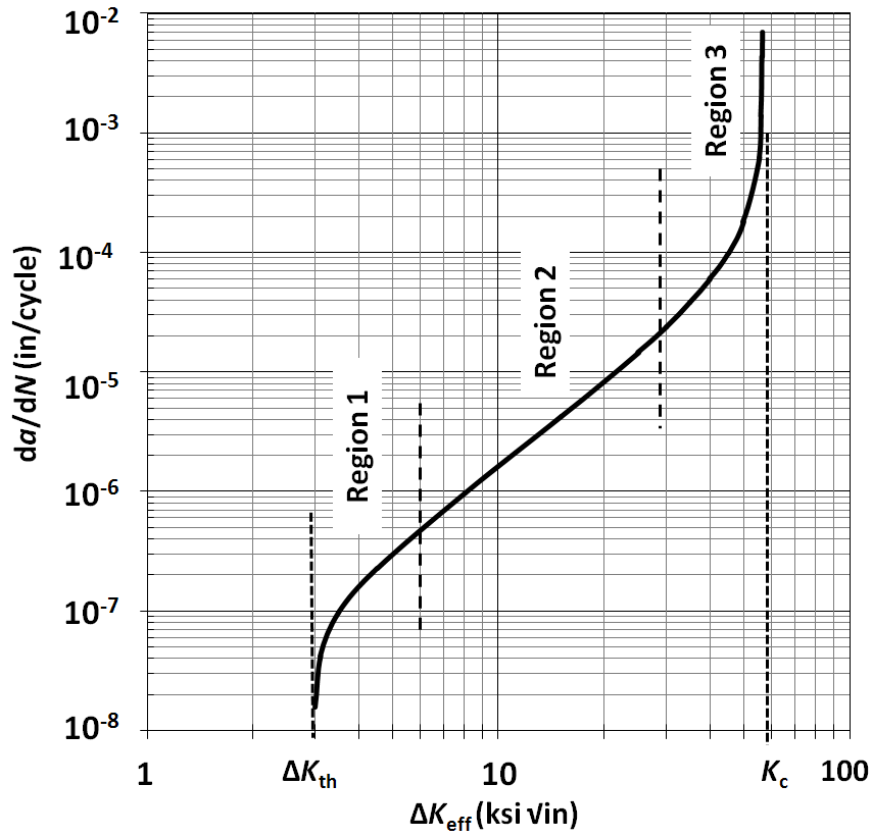


Figure 7.2: Idealized plot of crack growth as a function of stress cycles for a metal. By J. Hilgendorf, United Launch Alliance.

size. In addition, the fracture control program places restrictions on the materials that may be used and specifies the documentation needed to ensure that the correct material has been used, that it has been processed in a way to discourage the initiation of cracks, and that the proper inspections have been performed. It also specifies a factor to cover analysis uncertainty: typically, a fracture-critical part may be used for one-fourth of the life predicted by the safe-life analysis before it must be reinspected.

Because they involve inspection, fracture control programs are most commonly seen in aircraft and in reusable LVs such as the Space Shuttle. Expendable vehicles cannot be inspected after use unless they are recov-

7. Materials for Launch Vehicle Structures

ered, and then they will not be flown again anyway. However, expendable vehicles must undergo ground tests that consume some of the safe life of the parts, and inspection is possible after ground tests. So fracture control may be applied in expendable vehicle programs to a limited extent.

Specialized Materials

Most of the foregoing discussion applies to metals and composites, which are by far the most important materials used in launch vehicle structures. Their useful regime is linear elastic, and the effects of temperature and other environments on their behavior is small enough that it may usually be accounted for with adjustment factors. If a metal structure does yield, the amount of yielding is small enough that deformation plasticity in the form of an isotropic Mises yield function followed by a Ramberg-Osgood description of plastic flow, is usually sufficient.

For more complex materials such as elastomers, foam and adhesives, materials testing becomes even more expensive and time-consuming, and good property data accordingly harder to come by. Fortunately, these materials are often used in applications where very accurate mechanical property data are not vital. Many of these materials display time-dependent behaviors such as relaxation and creep, and have strong temperature dependence. They may also have nonlinear stress-strain curves, or may have such a large strain during operation that they must be treated with one of the many nonlinear theories of mechanics.

For materials that are not linear elastic, the distinction between phenomena and properties becomes important. Phenomena are behaviors such as elasticity, creep, and relaxation that can be observed and measured without assuming a particular material model. Observing material phenomena can be useful for screening or lot acceptance, and can *suggest* an appropriate material model, but are usually insufficient inputs for accurate simulation of structural response.

To conduct accurate analyses and simulations, a material model (constitutive equation) must be *assumed*, and only then can the properties defined in the model be measured. For instance, some type of stiffness may be measured for all elastic materials, but once one is forced to consider large strains of a compressible material, a large-strain model containing three properties may be necessary. A conventional uniaxial tension test will not suffice to determine the three properties; multiple

specialized tests are needed. A less desirable, but nevertheless common, approach is to adjust the properties until analysis agrees with a variety of measured responses that are similar to the actual application of the material.

Rational Methods of Materials Selection

Materials selection is a part of structural design optimization, whether the optimization is done intuitively by an experienced designer working on a minor variation of an existing design, or quantitatively through the use of a large material properties database and algorithms for adjusting hundreds of design variables.

The classical approach to optimum design, including material selection, was comprehensively reviewed in [19]. It involves the definition of a design index based on a requirement. For example, to optimize a thin-walled column, equations relating external load to the critical stress for two failure modes (column instability and local buckling) are derived, and by requiring that the margin of safety for both failure modes be minimized, a design index is determined. In this example, the index is a function of Young's modulus, some sort of plastic modulus, the load, and the length of the column. Given a set of values for some of these parameters, the others can be chosen so as to optimize the design index.

The design index approach is only tractable for problems involving a few key parameters. The ability to determine ahead of time which parameters are key is an aspect of engineering genius that not everyone enjoys. But by computerizing the process, the number of variables can be greatly enlarged, so an intuitive ability to narrow down the design space is less important. One such approach was documented by Mukhopadhyay [20]. Chapter 3 of the present book discusses materials selection in greater detail.

7.4 Structural Design and Requirements

Materials selection is as much a part of the design process as sizing. In fact, the two cannot be separated. Therefore, the requirements and criteria that impinge on the structural sizing process also impinge on materials selection.

7. Materials for Launch Vehicle Structures

Development practices in LV materials and structures are an interesting combination of extreme conservatism and bold risk-taking. Modern LV development programs typically budget for zero or one test flight before an expensive payload is launched. Differences in payloads and trajectories tend to limit the amount of knowledge that can be carried from one flight to the next. When a military service decides to launch a billion-dollar, one-of-a-kind payload critical to national security on an expendable LV in a configuration that may never have been flown before, the materials selection, structural sizing process, and testing are held to standards that owe more to custom than science. The launch decision itself is a major, irrevocable commitment of resources based on a significant extrapolation of experience. Therefore, the extrapolation process must be as rational as possible.

The following discussion is necessarily general, because program-specific policies are usually trade secret and/or export-controlled. This section does not purport to review unusual or innovative structural qualification methods, or specific reliability requirements, that are not documented in the public domain.

Contractual Requirements

By far the most significant requirements are those imposed by the procuring agency or, in the case of commercial operations, by the payload client. In some cases these requirements are actually drawn up by the LV contractor itself, subject to revision and approval by the procuring agency. Requirements exist in a hierarchy that is managed by systems engineers primarily to ensure that the LV delivers a functioning space vehicle to the desired orbit, and secondarily to minimize the cost, development time, danger to the public and other factors. The structural system, propulsion system, guidance and navigation system, and other systems are considered subsystems of the LV system as a whole. Blair and Ryan [21] provide a good overview of requirements and standards, and how detailed design criteria are derived from them.

A set of top-level functional requirements for the structural system that could well apply to many different LVs is

- to support and protect the other vehicle systems and the space vehicle such that they can function properly

- to contain and deliver working fluids to the propulsion, guidance and other systems
- to maintain an aerodynamically acceptable shape, and
- to do the above in a way consistent with the functioning of the other vehicle systems; for example by allowing electrical grounding.

Top-level requirements may specify not only the performance goals to be met, but also the *likelihood* that the design will meet them. It may be required that the vehicle be 98% likely to meet all requirements; that is, to place an intact payload into the proper orbit 49 out of 50 times on average.

It is difficult or impossible to predict whether a complex machine like an airplane or a launch vehicle will satisfy such a requirement *simply based on the design*. There are too many interacting failure modes. For aircraft, the large number of repeated operations makes it possible to develop some empirical rules of thumb. But even an empirical approach is usually not possible for LVs, because of the low numbers of identical vehicles and operations. Some researchers have attempted to use a Bayesian statistical approach to circumvent the lack of data [22]. An alternative might be to break the vehicle down into a few standard subsystems, and try to reuse those standard designs on many different vehicles, thus providing a significant experience base. But for LVs this is the exception rather than the rule.

Top-level reliability requirements are best interpreted as a general statement positioning the desired reliability relative to similar systems. It is healthy to realize that perfect reliability is neither possible nor desirable. For example, the Japanese space development agency set a reliability goal of 96% for their H-2A vehicle, stating honestly that they would *not* be “aiming for the ultimate in design” [23].

Laws and Regulations

The previous discussion covered requirements imposed by the procuring agency or self-imposed by LV contractors. Another class of requirements is that imposed by laws and regulations. These seek to minimize overflight and environmental hazards to the public. In the United States, the FAA regulates commercial space operations but not operations carried out by, or on behalf of, the federal government. This excludes the majority of

7. Materials for Launch Vehicle Structures

launches (and re-entries) from FAA scrutiny. Also, Title 14 of the Code of Federal Regulations does not impose the same very detailed structural requirements on LVs as it does on aircraft. It is mostly concerned with hazards from expended stages, re-entering payloads, and mishaps. The FAA's relationship with the private space launch industry is still evolving but it appears that private launches will not be regulated as closely as passenger aircraft. Therefore, vehicle safety laws and regulations do not significantly constrain materials selection for LV structures.

However, environmental regulations have had a significant and ongoing impact on materials selection for LVs, particularly in the area of coatings and insulation. Heavy metals such as cadmium, mercury and lead were once commonly used in metals processing and plating, but as it has become widely known that these substances are poisonous, regulations have greatly reduced their use. Beryllium has important aerospace structures applications due to its thermal stability, but beryllium dust is toxic and must be handled carefully. Also, the use of asbestos insulation and chlorofluorocarbon blowing agents for foam insulation has been greatly reduced by environmental regulations.

Range Safety

The other major class of requirements is that imposed by operators of launch ranges to minimize the risk of injury to personnel and damage to ground equipment. Military, government non-military and commercial organizations alike must adhere to range safety rules. The vast majority of LVs are operated out of the ranges listed in Table 7.2.

For many years, the governing range safety document for the Eastern and Western Ranges of the United States was EWR 127-1, *Eastern and Western Range Safety Policies and Procedures* [24]. Although EWR 127-1 states that it is "applicable to all organizations, agencies, companies and programs conducting or supporting operations on the ER and WR," it now only governs programs introduced at the Ranges prior to 2004. Since 2004, Air Force Space Command has issued the manuals AFSPCMAN 91-710, *Range Safety User Requirements Manual* [25] and AFSPCMAN 91-711, *Launch Safety Requirements for Air Force Space Command Organizations* [26] as replacements for EWR 127-1. The former is binding on all range users, but the latter is binding only on Air Force space programs.

EWR 127-1 sets as a general goal that the risk of injury or damage

Table 7.2: Major space launch ranges

Name	Launch Location(s)	Notes
Eastern Range	Cape Canaveral Air Force Station, Florida	Mainly low-inclination orbital vehicles on a southeastward ground track
Western Range	Vandenberg Air Force Base, California	High-inclination orbital vehicles on a southward ground track, and suborbital vehicles westward toward Kwajalein Atoll
Wallops Research Range	Wallops Island, Virginia	Small suborbital and orbital vehicles in eastward to southward directions
Guiana Space Centre	Kourou, French Guiana	Orbital vehicles to a wide range of inclinations
Baikonur Cosmodrome	Tyura-Tam, Kazakhstan	Orbital vehicles along a corridor extending northeastward over Russian territory
Plesetsk Cosmodrome	Arkhangelskt Oblast, Russia	Northward into high-inclination and polar orbits
Sea Launch	Equatorial Pacific Ocean	Low-inclination orbital launches

7. Materials for Launch Vehicle Structures

to the public due to space launches should be no greater than that normally accepted in day-to-day activities, including the risk due to airplane overflights. It uses language such as “all reasonable precautions shall be taken” and “lowest risk possible”.

Section 3.12 of EWR 127-1 contains detailed requirements for testing and analysis of pressurized systems and structures on LVs.³ It requires that materials be compatible with working fluids, seals, lubricants, and so on, from the standpoint of flammability, ignition and combustion, toxicity and corrosion, and requires the range user to supply evidence in the form of a report. It specifies that material compatibility should be based on T.O. 00-25-223, *Integrated Pressure Systems and Components (Portable and Installed)*, Chemical Propulsion Information Agency Publication 394 [27], MSFC-HDBK-527 [14], or independent testing.

EWR 127-1 also specifies qualification, acceptance, hydrostatic proof and leak testing requirements for pressure vessels and pressurized systems. It requires quite specific design solutions to reduce risk, such as the location of drains and vents, design of interconnects, and the like. It addresses graphite-epoxy composite overwrapped pressure vessels (COPVs) in a separate appendix, which requires demonstration of a leak-before-burst (LBB) failure mode for metal-lined COPVs, non-destructive evaluation of the composite overwrap, special fluid compatibility testing, and design/test/pedigree record-keeping in accordance with MIL-STD-1522 [28]. These requirements are for the safety of ground personnel and the public. For small-diameter lines in particular, static design factors may be as high as 4.0 and required safe-life may be as long as four expected service lives.

The very detailed and prescriptive regulations in EWR 127-1 were consciously relaxed in the new AFSPC manuals, not necessarily with the intention of raising risk, but rather to change the approach from risk avoidance to risk management. Some specific materials selection rules in EWR 127-1 have been deleted from the new manuals. The thinking behind this is outlined in a National Academy of Engineering study [29]. Quantitative requirements have replaced the “all reasonable precautions”

³As defined by EWR 127-1, a pressurized system is a system such as a helium storage bottle that is primarily designed to contain internal pressure, while a pressurized structure is a system such as a main propellant tank that carries both internal pressure and significant external loads.

language, and the range user is given more discretion in implementation. This initiative was partly driven by the desire to reduce the cost of range safety and make the ranges more attractive to commercial users.

Verification and Qualification

A vehicle can meet all design requirements but still fail to deliver the payload to orbit. Further, because of randomness in material properties, dimensions and loads, one successful flight of a system does not guarantee future flights will also succeed. Even in the case of a reusable vehicle, 49 successful flights do not verify the requirements are met if the design lifetime is 50 flights. The vital question, and one that the materials and structures engineers must help answer, is whether the *next* flight will be successful.

Analysis and review of ground test and previous flight data are necessary, bearing in mind that predictions of future flight performance are at best a *rational extrapolation* of experience. The benchmarks determining whether the system is ready for the next flight are set cooperatively by the materials and structures engineers, the systems engineers, and others. Some engineers, notably Sarafin [30] in reference to satellite structures, refer to these benchmarks as verification criteria rather than requirements. The distinction is made in order to discourage blind adherence to rules, because after all, those criteria only represent an educated guess as to the best way to build confidence in system reliability.

The overall means of qualifying LV structural hardware for flight may be a contractual mandate, a company policy, or simply tradition, but the preferred method of qualifying launch vehicle primary structure will always be a single-article test to limit load times a factor. Other verifications such as proof testing or analysis are adjuncts to this basic approach.

Requirements can be so narrowly written that they are really prescribed designs that hold back the state of the art. It would not be desirable, for instance, to require propellant tanks to be designed to a one-size-fits-all specification such as that used for rivets. But though excessively narrow requirements and standards may have been imposed in areas such as avionics, this was not the case in materials and structures. The United States Department of Defense abandoned military standards and even prohibited contracts from citing them as requirements for a time

7. Materials for Launch Vehicle Structures

in the 1990s [31]. This was part of a government-wide political initiative that affected NASA (“better, faster, cheaper”) as well [32]. After a series of high-profile failures in the late 1990s, procuring agencies concluded that wholesale abandonment of standards was too extreme, and systems engineering processes began to reintroduce them [33].

Structural Qualification

In this section, the most commonly used concepts in structural qualification are introduced. While terminology varies, these concepts appear in most government standards concerning structures, and knowing their meaning is a prerequisite to understanding the various qualification strategies.

Design limit load is the maximum expected in-service load. Programs may be very precise; a common definition is that limit load is the 99.7 percentile of a distribution of loads that may be generated by analysis, flight measurements, or both. Such loads are usually generated from a finite number of samples, so it is often stated additionally that the 99.7 percentile load must be determined to a confidence level of 90%.

Design factors are factors by which limit load is multiplied to determine the *no-yield condition* (the load at which the structure must not suffer detrimental deformation), the *proof condition* (a load used for acceptance testing), the *no-break condition* (the load at which a structure must not lose its load-carrying capacity, through breakage or instability), and other hypothetical load levels used in analysis. Design factors are chosen by, or subject to the approval of, the procuring agency.

Test factors are analogous to design factors but are used to factor up the limit load for testing purposes, as opposed to design purposes. They are usually equal to the corresponding design factors, but they do not have to be. For example, if limit load is 10 tons, and the design ultimate factor is 1.25, analysis must show that the structure will withstand a load of 12.5 tons. Most likely an ultimate load test would also specify a load of 12.5 tons, but it could specify 14 tons or some other factored-up value.

Since limit load already takes quantifiable uncertainties into account, design and test factors can be viewed as insurance against “unknown unknowns.”

Capability is a lower bound on the ability of a structure to resist detrimental deformation and to maintain its load-carrying capacity. It is

determined by analysis using material yield and ultimate strengths (which are lower-bound values) and the least favorable dimensions allowable in built hardware.

Margin of safety or simply *margin*, is the fraction by which the capability exceeds the no-yield or no-break conditions. Thus, continuing the example above, if the structure is predicted to buckle or break at a load of 15 tons, the ultimate margin would be

$$15/(1.25 \times 10) - 1 = +22\% \quad (7.1)$$

The sign is customarily shown on a margin even if it is positive. Using this definition, the capability may be viewed as the load at which the margin of safety is zero.

The *demonstrated load* is the load by which the test factors were multiplied in generating loads during a successful test. Generally, there are two tiers of design factors: a lower set of values, meant for use on structures that have been tested, and a higher set, meant for use on structures that have not been tested. To be entitled to use the lower, “tested” set of design factors, a structure cannot be exposed to flight loads in excess of the demonstrated test load. In such situations, the demonstrated load becomes the *allowable load* for the structure. Even if the margin is positive at the allowable load, flight loads must not exceed it, otherwise the lower design factor is no longer justified.

The demonstrated load is sometimes known as the limit test load, and the demonstrated load times the ultimate test factor is sometimes known as the ultimate test load. However, these should not be confused with design limit and ultimate conditions. The test loads are fixed once the test has been completed, but the design conditions may vary as knowledge is gained about the LV.

For an untested structure, the allowable load is the load at which the margin of safety is zero. In other words, for an untested structure, the allowable load equals the full capability. In contrast, large test articles are not usually tested to full capability or to destruction, only to design limit load or less, thus constraining the flight article to an allowable load at which ample margin may exist. The “hidden margin” between the allowable load and the capability of a tested structure is an important fact to consider when comparing the relative risk of testing versus not testing a structure. Testing can uncover a dangerous condition that analysis

7. Materials for Launch Vehicle Structures

alone might miss, even when higher safety factors are used to compensate for the lack of testing.

The relationship between the various design conditions, the test and analysis results, and the design factors and margins is illustrated in Figure 7.3. This figure shows the predicted flight loads and predicted failure loads in the form of histograms, which could be generated by Monte Carlo simulations or from an assumed distribution. For instance, an individual failure load might be calculated Monte Carlo-style from random draws of material strength and dimensions from distributions consistent with sampled test and dimensional data. Or, more commonly, it may simply be a Gaussian distribution fit to a mean and variance. Flight loads are more likely than failure loads to be built up from random underlying contributors, but in principle both can be done that way.

The figure shows the capability as a lower limit on predicted failure loads, and the design limit load as an upper bound on predicted flight loads. The illustration shows the typical circumstance in which flight load predictions are more scattered than failure load predictions. This arises from greater underlying uncertainty in wind statistics, trajectories, and other inputs to the loads analysis, as well as uncertainty in the analytical model itself. It also shows that the capability and limit load do not enclose every single predicted load, and in that sense they are not truly bounding values although we call them that for convenience.

The demonstrated limit load is a single value, shown in gray on the figure. It is typically close to the design limit load. The intent is usually to test the structure to exactly the limit load, but limit load can change as new knowledge is gained. Finally, the figure shows that the design factor provides separation between limit load and the no-fail condition, and the separation between limit load and the capability is a function of both the design factor and the margin of safety.

One may hear a statement like the following: “The test article was loaded to 140% of the no-yield condition, so a tested margin of 40% has been established.” This is not a correct use of the term margin, because the test was of a single article that could have been stronger than average. Margins are based on lower-bound strength, not averages. It would, however, be correct to say, “The test load was 90% of capability, so there was a 10% margin of safety during the test.” The capability represents the lower-bound strength, and the test load is known, so there is no need

to account for uncertainty in the load. Therefore, the stated margin of 10% is meaningful.

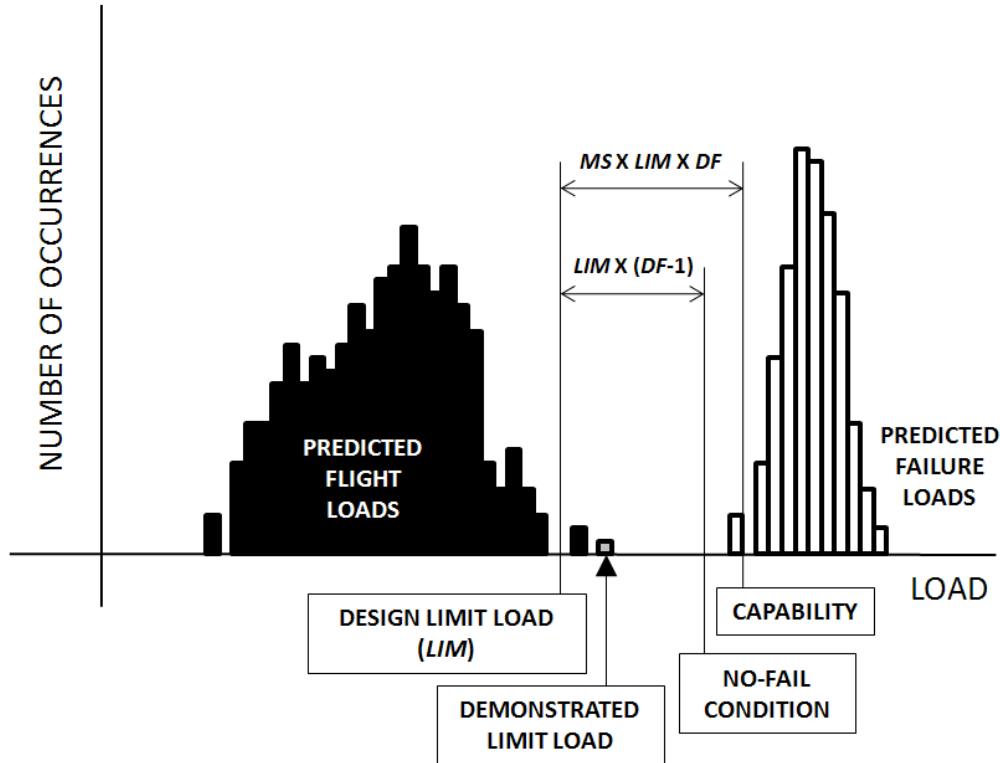


Figure 7.3: Distributions of predicted loads, failure loads, and the separation of the two provided by lower-bound capability, upper-bound limit load (LIM), chosen design factor (DF), and realized margin of safety (MS). By the author.

Stiffness is as important as strength in LVs. The thin-walled construction, combined with the strength and stiffness properties of typical materials, tends to render the buckling margins about the same as the strength margins, and both are always checked. From a material properties standpoint, stiffness is less variable than strength and is less affected by temperature and moisture. Therefore, nominal modulus values are often sufficient, especially for metals.

In composites, a lower-bound stiffness may be obtained by testing “hot-wet” samples; that is, coupons saturated with moisture and held at the maximum expected service temperature. But composite stiffness

7. Materials for Launch Vehicle Structures

properties that are truly applicable at the scale of a full structure can be challenging to measure. Specially laid up and cured coupons may have different microstructure than the full-scale component. Coupons cut out of a full structure may have damaged edges.

However, analysis for stiffness is less exact, and therefore more conservative, than analysis for strength. The buckling failure mode is the one most influenced by stiffness. Because the buckling load of a thin-walled shell is strongly affected by slight geometric imperfections and edge constraint, adjustment or “knockdown” factors derived from experiments on subscale specimens are applied. These factors may lead to a reduction in the predicted buckling strength of 50% or more, as compared to the theoretical value for a geometrically perfect shell. Factors documented in a NASA monograph [34] were originally developed from experiments on small plastic cylinders. Bushnell comprehensively reviewed the state of the art in shell buckling analysis through 1980 [35]. Recently, recognizing the major role played by buckling knockdown factors in vehicle design, NASA conducted a Shell Buckling Knockdown Factors research program that was the most significant work in the field in decades and which experimentally supported a significant refinement and reduction in the factors [36].

Pitfalls, Controversies and Engineering Judgment

Stated requirements, and the strictness with which they are applied, vary between programs. Knowing what to require in a particular situation depends largely on factors specific to each program. Such factors are neither public nor readily transferable to new situations, so this discussion is limited to the pros and cons rather than advocacy of particular solutions. Because primary structure must be qualification-tested to the no-break condition, if predicted loads increase, for instance due to payload weight growth, analysis refinements, correction of mistakes, and so on, the structure must be retested. However, an expensive and time-consuming retest will only be contemplated if the increase in loads is “significant.” There may be special provisions for allowing higher loads on a structure than what it was tested to, possibly using a sliding scale of design factors.

Also, it is sometimes not easy to determine the range of applicability of a structural test. If a material must be slightly changed from that used in the test article, is the design with the new material still qualified, or

must it be re-tested? From this scenario comes the idea of qualification by similarity. This refers to a formal process of demonstrating that a design may be considered test-qualified even though it is not identical to the test article. A detailed comparison of material, geometry, and manufacturing differences is necessary, as defined in MIL-HDBK-340 [37]. An example of qualification by similarity occurs when a propellant tank must be enlarged to meet new mission requirements. The course usually followed is to “stretch” an existing, qualified design. Often, the stretched design may be considered test-qualified, even though it is longer than the original test article. The guiding requirement in such cases is that the new design must have the same failure modes as the original, with equal or higher margins of safety.

There is controversy in the definition of primary and secondary structure and its implications for testing. The fundamental divergence may be illustrated by considering two structures, A and B. Suppose Structure A was successfully qualification-tested and has zero margin of safety using tested design factors. Structure B was not qualification-tested but has zero margin of safety using higher, no-test design factors. May the two structures be considered equally acceptable under all circumstances?

One school of thought says that the reliability added by using the higher, no-test design factors completely compensates for the lack of testing. Using typical values, consider that a structure with zero ultimate margin using a design factor of 1.60 would have a margin of 60% if a design factor of 1.00 were used. From this perspective, a program may elect not to test some primary structures. The distinction between primary and secondary is then made mostly on the basis of size: the vehicle can tolerate “fat” designs of small structures needed to accommodate no-test design factors, but cannot tolerate fat designs of larger structures. Therefore, larger structures are tested only to enable the use of lower, tested design factors. This less conservative viewpoint is characteristic of programs without heavy involvement of a procuring government agency.

The other school of thought posits that higher safety factors can never completely compensate for the risk of an analysis shortcoming that would only be revealed by testing. Therefore, primary (critical, non-redundant) structure *must* be qualification-tested, whether or not it has positive margins using no-test design factors.

Also important is the “hidden margin” discussed previously. There is a

7. Materials for Launch Vehicle Structures

long history of success in operating structures qualified by test, but those structures were usually neither tested nor flown to their full capabilities. Structures qualified solely by analysis, using a typical no-test design yield factor of 1.60, have allowable loads $1 - (1/1.60) = 38\%$ lower than capability. But successfully flown, tested structures were most likely limited to loads 20% lower than capability simply because they were not tested to full capability and were limited in flight to the test-demonstrated load. Therefore, allowable loads for non-tested structures would be not 38% lower than the experience base, but rather only about 18% lower. The “pad” provided by the no-test factors of safety does not appear quite so comfortable when viewed this way.

Structures that are nearly always considered primary are:

- Fairings
- Payload fittings and adapters
- Main propellant tanks
- Interstages, intertanks, skirts and transition sections
- Engine thrust structures

Outlook

The level of conservatism that ultimately proves more cost-effective is different in every case and is what makes structures engineering more than just a calculation process. It is not surprising that the organization that bears the cost of testing tends to take a less conservative approach, whereas the organization that bears the cost of a failed mission tends to be more conservative. When the same organization bears the costs of both testing and flight failures, a rational ordering of priorities is forced. But often, the responsibilities are separated, and the negotiated level of conservatism is determined by a political process, not an objective technical one.

Current flight rates are too low to conclusively prove which approaches are superior. The structural subsystem itself, and especially any single structure, must have a very remote chance of failure in order for the vehicle as a whole to have a reasonably small (say, one in a hundred) chance of failure. It is not uncommon for the required probability of failure for a particular structure to be on the order of one in a million. Even if a less conservative approach leads to double the chance of failure (say, 2×10^{-6})

for a single structure, this will not be empirically distinguishable from a more conservative approach over the life of a program. The danger is carrying this thinking over, by inattentive systems engineering or lax verification of requirements, to *every* structure. Then, of course, the vehicle as a whole will have twice the risk of failure.

It has been noted that a truly reusable LV would allow requirements to be made more rational, as they are in aeronautics, by generating a large performance database for the same flight article.

A look at launch vehicle failure statistics shows that the overall demonstrated reliability of LVs worldwide was 96% for the period 1984-1994 [6]. Of the failures, the propulsion system was by far the leading cause (27 out of 43 failures).⁴ Just five out of 43 failures were attributable to primary structure in that period: a payload fairing failure on a Chinese CZ-2E, a Centaur liquid oxygen tank failure, and three solid rocket motor case failures, including the well-known *Challenger* disaster. Many failures cause the vehicle structure to be destroyed, but these are usually due to primary failures in other systems leading to loads in excess of those the structure was designed to sustain. In such scenarios, the structure is not considered the root cause of failure.

A probabilistic approach to structural integrity would dispense with the question of primary versus secondary structure. Instead of using design factors, in a probabilistic approach, each component would be assigned a probability of failure considering all sources of uncertainty.

7.5 Pressurized Structure

The majority of material in a space launch vehicle is found in integral load-bearing propellant tanks. This section is mostly confined to discussion of materials for the tank shells; tanks also have small parts such as sumps, lids, and outlets that are subject to different requirements than the shells.

Propellant tanks function as pressure vessels, containing fluids under moderate pressure and often at cryogenic temperatures. However, unlike stationary pressure vessels, propellant tanks must sustain large, highly variable primary flight loads. This has been the case since the early days of rocketry, when for reasons of weight, external load-bearing shells protecting tanks from flight loads (as in the V-2) were replaced by

⁴I have counted solid rocket motor case failures as structures failures.

7. Materials for Launch Vehicle Structures

integral load-bearing tanks. Also, the need to reduce mass has required that propellant tanks be much more lightly constructed, with far smaller design factors than stationary pressure vessels. Finally, propellant tanks in expendable vehicles are operated for only a short time, so long-term, time-dependent processes such as creep and corrosion are less relevant. Flynn, in a book covering all aspects of cryogenic engineering, devotes some discussion of propellant tanks as compared to other applications of cryogenic technology [38]. He also provides a useful discussion of cryogenic insulation, which will be discussed later in this chapter.

Government standards such as range safety requirements consider the main propellant tanks to be “pressurized structures” rather than pressure vessels (refer to [24] for one formal definition), reserving the designation of pressure vessel for smaller tanks such as propulsion system pressurization tanks that do not bear significant external loads. Factors of safety and other requirements are much different for pressurized structures, as opposed to pressure vessels.

Propellant tanks are of three basic designs. The commonest is the stiffened metal shell, structurally stable under the load of its own weight when empty and unpressurized. Stiffening is generally by integrally machined stiffeners in an isogrid or orthogrid pattern, rather than by mechanically fastened stringers. Such designs are constructed of aluminum alloys. The next most common is the “steel balloon” design, which is very thin-walled and not structurally stable under the load of its own weight unless pressurized or stretched. Its stability before fill and pressurization is maintained by pressurization with an inert gas or by mechanical tension applied by a holding cradle. This design was most famously applied in the Atlas missile.

Both the stiffened and balloon-style metal designs may be of a single tank space, containing either fuel or oxidizer, or combined fuel and oxidizer tanks separated by a common, dome-shaped internal bulkhead. The common-bulkhead tank offers mass and size savings over separated fuel and oxidizer tanks, and has been used in such high-performance upper stages as the Saturn S-II and S-IVB [39] and the Centaur. A drawback of this design is the need for the common bulkhead to control heat flow between two propellants that may be at vastly different temperatures.

The third type of design is the composite tank. Whereas non-cylindrical shapes would be very difficult to achieve in a mass-efficient

manner with metallic shell designs, such shapes are less troublesome with composites. Also, composite tanks offer potentially significant mass savings through higher material specific strength and the ability to orient the primary load-carrying direction of a composite laminate along the expected loading direction. Composites also offer better resistance to fatigue and flaw propagation, because microscopic flaws tend to be blunted and stopped by the fibrous microstructure, although accumulated fatigue damage can result in increased permeation of propellant. With all these advantages, much effort has been expended on realizing an operational composite propellant tank, but to date, successes have been small in number.

All tank designs must perform the basic function of containing the liquid propellants during testing, fueling and flight. Propellants vary from RP-1, a highly refined kerosene, to cryogenic liquid oxygen (LOX) and liquid hydrogen (LH₂), to storable but often toxic combinations such as hydrazine and nitrogen tetroxide. All have properties that constrain the designer's choice of propellant tank materials, and cryogenic propellants require that the tank be insulated to minimize boil-off.

In almost all cases, tanks must sustain aerodynamic and inertial flight loads, which for the typical long, cylindrical tank means a combination of axial compression and bending. The Space Shuttle external LOX tank is a special case in that it receives axial aerodynamic loading directly due to its position at the forward end of the tank assembly. Inside the tank, various baffles and propellant management devices must be supported. Finally, depending on the tank's location in the vehicle, main propellant feedlines and electrical tunnels must be supported, either as an external appendage or through centerline tunnels as in the Saturn S-IC stage [39].

The tank contents must be fed to the engines under pressure. For a pressure-fed propulsion system, propellants are forced directly into the combustion chamber by ullage pressure. The ullage is the unfilled space at the forward end of the tank. For pump-fed engines, moderate pressure is still necessary in order to prevent cavitation in feedlines. Just prior to launch, large tanks are pressurized using a ground supply of gas; once the booster engines have been started, the gas supply may be provided by the engines through a re-pressurization system. For smaller stages, an onboard supply of inert pressurant is often used.

It is worth recalling the basic relationship between load and internal

7. Materials for Launch Vehicle Structures

forces for a pressurized thin-walled cylinder subject to an external compressive force P_0 and bending moment M_0 at the tank bottom. (Shear force is usually not significant when considering the overall section forces acting on a launch vehicle.) For increased generality, suppose the cylinder contains a quantity of liquid of density ρ and that it is accelerating forward at a rate a .

The tank and its contents together do not form a continuous elastic body, so they must be analyzed separately. The pressure at a distance z below the free surface of the liquid is

$$p(z) = \rho z(a + g) + p_{\text{ull}}, \quad (7.2)$$

where p_{ull} is the ullage pressure and g is the acceleration of gravity.

A separate free-body diagram shows that the axial compressive force in the tank shell at location z is

$$\bar{P}(z) = P_0 - m(z)a - \pi R^2 \rho h(a + g) - \pi R^2 p_{\text{ull}}, \quad (7.3)$$

where h is the total height of the liquid in the tank, R is the tank radius and $m(z)$ is the mass of the tank aft of location z . Part or all of the force $P = P_0 - m(z)a - \pi R^2 \rho h(a + g)$ may be provided by a separate loads analysis. It may include, additionally, vibratory effects and other terms not shown in this simple analysis. Consider the typical case where the force is given in the form

$$\bar{P}(z) = P - \pi R^2 p_{\text{ull}}. \quad (7.4)$$

The bending moment at all locations, assuming for simplicity no lateral forces or angular acceleration, is $M = M_0$.

Bending stresses due to the moment load M are calculated as though the tank were a slender, hollow beam of wall thickness t . The longitudinal stress has its maximum (highest tensile) value at one of the two points on the cross section farthest from the bending axis, and its minimum (highest compressive) value at the other such point. The largest longitudinal compressive stress is

$$\sigma_{z,\text{comp}}(z) = -\frac{\bar{P}(z)}{2\pi R t} - \frac{M}{\pi R^2 t} \quad (7.5)$$

$$= -\frac{P}{2\pi R t} - \frac{M}{\pi R^2 t} + \frac{p_{\text{ull}} R}{2t} \quad (7.6)$$

and the largest longitudinal tensile stress is

$$\sigma_{z,\text{tens}}(z) = -\frac{P}{2\pi R t} + \frac{M}{\pi R^2 t} + \frac{p_{\text{ull}} R}{2t} \quad (7.7)$$

The hoop stress is

$$\sigma_{\theta}(z) = \frac{[\rho_z(a+g) + p_{\text{ull}}]R}{t} \quad (7.8)$$

The quantities

$$P_{+\text{eq}} = P + 2M/R \quad \text{and} \quad P_{-\text{eq}} = P - 2M/R \quad (7.9)$$

are called *equivalent axial loads* [40], and in terms of them the longitudinal stresses are

$$\sigma_{z,\text{comp}}(z) = -\frac{P_{+\text{eq}}}{2\pi R t} + \frac{p_{\text{ull}} R}{2t} \quad (7.10)$$

$$\sigma_{z,\text{tens}}(z) = -\frac{P_{-\text{eq}}}{2\pi R t} + \frac{p_{\text{ull}} R}{2t} \quad (7.11)$$

In the preceding σ represents the average stress over the wall thickness. Often, a local analysis that considers the variation of stress between the skin and the stringers or the core and face sheets of a built up wall needed. In such cases it is useful to work in terms of q , the integral of stress over the wall thickness:

$$q_{(z,\text{comp})} = -\frac{P_{+\text{eq}}}{2\pi R} + \frac{p_{\text{ull}} R}{2} \quad (7.12)$$

$$q_{(z,\text{tens})} = -\frac{P_{-\text{eq}}}{2\pi R} + \frac{p_{\text{ull}} R}{2} \quad (7.13)$$

$$q_{\theta} = [\rho_z(a - g) + p_{\text{ull}}]R \quad (7.14)$$

The quantity q is called the line load or tensile flux. Note that in all of the above development, axial force is taken as positive in compression.

These equations apply to large tanks and cylindrical adapters except where local irregularities or constraints render the underlying assumptions invalid. For a structure such as an adapter or interstage that contains no liquid, the terms containing density may be deleted. However, internal pressure in such structures may be important. Consider that an adapter

7. Materials for Launch Vehicle Structures

with a radius of 100 inches and a wall thickness of 0.2 inches will experience a longitudinal wall stress of 0.25 ksi for every psi of internal pressure.

From Equations (7.8), (7.10) and (7.11), we see that in the absence of external load and static head, the state of stress in the membrane is biaxial with a hoop-to-longitudinal ratio of two. External loads will cause this ratio to vary significantly from two. Conventionally, material strength is determined from uniaxial tensile tests, and then a combined-stress yield theory such as the Mises theory is used to calculate a scalar effective stress from the actual biaxial state of stress in the application. Although a large amount of experimental effort has been directed toward gaining a more sophisticated understanding of metal yielding and rupture under biaxial stresses (see [41] for example), the results seem to be little used today.

The use of the maximum principal stress failure criterion for metals is near-universal, but consider that a ductile material has a higher ultimate stress than its strength at rupture. In fact, for some high-strength steels, the stress is higher at the offset yield point than at any subsequent time [41]. Though maximum principal stress correlates very well to rupture strength, it is possible that ultimate stress, which is the material property customarily used to indicate failure, might be predicted better by alternative criteria.

The foregoing discussion only addresses strength. Tanks may also fail by global or local buckling, or by the fracture of a flaw at far-field stresses below yield. In practice, the margin of safety tends to be about the same for strength and buckling failures. The fracture failure mode, which is managed by controlling the initial flaw size, may not be close to the others in criticality.

Proof pressure testing is usually required, if not by the procuring agency, then by the range safety organization. Pressure testing at cryogenic temperatures is very expensive, so proof testing is usually done with room-temperature nitrogen gas or water. The ratio of yield to ultimate strength, and the fracture toughness, of many materials is different at room temperature than at the service temperature. Thus, it is not a trivial problem to devise a room-temperature proof test that exercises all failure modes of a cryogenic propellant tank adequately.

Designing for light weight requires that the structure be quite thin-walled. Thicknesses (or effective thicknesses, in the case of stiffened

structure) can be on the order of one tenth of an inch for a section 200 inches in diameter ($R/t = 2000$). For comparison, a soda-pop can has $R/t \approx 1000$.

Methods of flaw screening over large areas are usually sensitive enough to allow very small initial flaws to be assumed in the safe-life analysis and thus to provide ample safe life.⁵ Automation of flaw screening can be developed during production planning. Years ago, flaw screening was provided via proof test; a flaw that could survive the proof test without catastrophic propagation was considered very likely to survive flight as well. This was usually performed on pressure vessels, and pressurized structures. A more rigorous screen may (depending on the material) be provided by a proof test at cryogenic temperatures. For many materials, at colder temperatures the yield strength increases, permitting testing to a higher pressure, and the fracture toughness decreases, reducing the margin against catastrophic flaw growth.

Methods of flaw detection include dye penetrants, ultrasound, x-ray, magnetic particles, and eddy current inspection. The inspection method is chosen based on cost, the required sensitivity, the accessibility of the area to be inspected, surface finish and coating/plating, and the material. MSFC-STD-1249 [42] is an oft-cited standard covering inspection methods.

Some materials have high fracture toughness relative to yield strength, so a larger flaw can be tolerated. The ratio of toughness to yield strength is significant due to the need to restrict stress levels below yield strength. Conversely, a high-yield-strength material with low fracture toughness will need to be screened for very small flaws, which is the case with some high strength steels with low ductility.

In some cases, when hardware is received, it is found to have been inadequately inspected, or the results of the inspection may show that the design intent was not met. It may prove faster and cheaper to conduct additional analysis, inspection and testing to accept the discrepant hardware than to scrap the structure and manufacture a new one.

⁵This discussion provided by John Hilgendorf, Structural Analysis Lead for Delta II, United Launch Alliance.

The Leak-Before-Burst Criterion

It is usually required that pressure vessels and pressurized structures satisfy the “leak-before-burst” (LBB) criterion. The LBB concept is found in all industries that use pressure vessels, including aerospace, energy, and ground transportation. A definition given in a commonly cited military standard [28] is “a fracture mechanics concept in which it is shown that any initial flaw will grow through the wall of a pressure vessel and cause leakage rather than burst.” The purpose is to prevent catastrophic or explosive failures of pressure vessels or pressurized structures that may damage nearby flight hardware and launch facilities, or injure personnel. In flight, a tank that has the LBB property may fail gradually enough that the mission can still be completed. It also provides time to depressurize or safe the system once a detectable leak has occurred.

If all pressure vessels in a system are held to the LBB standard, safety rules and nearby systems need not be designed to withstand an explosive, catastrophic failure. This saves money.

The LBB property may be verified by testing, analysis, or a combination of the two. A burst test that results in gradual leakage rather than sudden rupture is a demonstration of LBB. However, a test of a single article is of limited use unless it can be shown that an initial flaw not obviously detectable existed in a critical location. Analysis is necessary to determine the worst-case location and orientation. Flaws may be intentionally introduced into the test article to cause leakage to occur first at a location of interest.

Analytically, to demonstrate LBB, it must be shown that the vessel can withstand the expected operating pressure when a leaking (through-wall) flaw exists. Said differently, a crack growth analysis must show that the critical flaw size is larger than the wall thickness.

Stable Metal Tanks

Structurally stable metal tanks are the most common design. Historically, 2000-series aluminum alloy has been by far the most popular material in this application, although recently, lighter aluminum-lithium (Al-Li) alloys have been used. For relatively slender tanks, the cylindrical tank barrel may be formed as a single ring if small enough, but more commonly it is built up from panels. The end domes are usually spun and may be

of a different temper from the barrel. The barrel and domes are joined by welding. Squat tanks such as the S-II stage LOX tank have been laid out completely as domes welded from gores, with no cylindrical section. Large domes may be produced by explosive forming, as in the S-II stage [39]. Mynors and Zhang [43] discussed the widespread use of explosive forming in the 1970s, detailed the advantages and disadvantages, and described a research program exploring potential modern applications. Small end closures may be present at the apex of the domes, and these are usually bolted on so that they may be removed if necessary.

Barrel panels are stiffened either with extruded stringers or with integrally machined stiffeners. The integrally machined designs demand that plate be available in fairly thick gauges (one inch or more). Stiffeners may be created by machining or chemically milling pockets into a thick plate. The machining process leaves thickened weld lands, which are necessary because welds are not as strong as the as-machined metal. Machining of stiffeners is conducted when the panel is still flat, as a rule. Once machined, the panels are bump-formed or brake-formed into cylindrical arcs and then welded into a barrel of circular cross section. To avoid local buckling of ribs during forming, the machined pockets may be filled with a thermoplastic compound, then round the panel after the compound has cooled and hardened. The hardened compound provides stability to the thin ribs. After forming, the compound is melted out [44].

Because of the large amount of material that is removed, integrally machining the stiffeners may result in a scrap ratio of as much as 80%. This can be a significant cost for the more expensive alloys, and has been a motivation to attempt to produce Al-Li panels with extruded rather than machined stiffeners [45].

The isogrid pattern [46], in which the integral stiffeners are a network of equilateral triangles, is by far the most popular of the integrally stiffened tank wall designs. It offers the stiffness and mass efficiency of other stiffener patterns but preserves the large-scale isotropic behavior of the panels, so that they may be modeled as shells with “equivalent isotropic” properties. While the simplifications made possible by isotropic behavior may not appear to be very advantageous in detailed stress analysis, when rapid iterations must be done in design trade studies, isotropic behavior is a significant benefit. Meyer *et al.* [46] provided the definitive work on isogrid design and stress analysis.

7. Materials for Launch Vehicle Structures

The stiffening of tank walls by integrally machined stiffeners increases both the extensional and the bending stiffness of the walls. This improves the buckling (particularly the local buckling) resistance and the ability to withstand concentrated loads perpendicular to the shell surface at openings or attachments. The same principle is followed for structures other than tanks, where integral ribs or mechanically attached stringers, corrugation, or sandwich construction may be used.

Propellant tank barrels and domes are invariably joined by welding, but welding is challenging in this application because of the relatively thin material and the tapered thicknesses that are used to save weight. Weld schedule development is time-consuming and external support is usually necessary to avoid distorting the shell due to the required heat input. Mendez and Eagar [47] provide an overview of the state of the art in aerospace welding technology; a more detailed discussion is presented in the section on manufacturing later in this chapter.

The 2000 series of aluminum alloys has historically been the material of choice for stable tank designs and remains dominant, although in the last 10-15 years, the Al-Li alloys have also become significant. Chapter 2 covers aluminum alloys in detail. The 2000-series alloys are aluminum-copper alloys with the percent of copper varying from 0.9% to 6.3%. In these alloys, the intermetallic compound CuAl_2 serves as the primary strengthening ingredient. Silicon and lithium are added to allow room-temperature age hardening, as well as improve the forgeability and strength. Trace amounts of manganese, magnesium and titanium are present to refine the grain and inhibit stress corrosion [48]. Alloys for tank applications must be weldable, so that large barrels can be built up from smaller panels, and their strengths must be insensitive to notching at cryogenic temperatures. The 2000-series alloys were the highest-strength weldable alloys available for many years. Higher-strength alloys such as the 7000-series are available, but their poor weldability and cryogenic notch toughness relegates them to use in interstages, where they are assembled using fasteners and not subject to extremely low temperatures [49].

A very popular tank material is Alloy 2219, a high-strength, weldable aluminum alloy whose principal alloying element is copper (6.3%) [50]. It has been the primary tank structural material in the Saturn S-IC stage [49], and the standard-weight and lightweight (LWT) Space Shuttle

External Tank designs [51].

Alloy 2219 is a wrought, heat-treatable, precipitation-hardening alloy developed by Alcoa in 1954 for high-temperature structural applications [50]. However, its excellent properties at cryogenic temperatures are what makes it attractive for LV tanks. Its full strength is developed by solution heat treatment followed by aging. Cold work may be applied before aging to further enhance the precipitation hardening process. Reheat of clad grades (not commonly used in LVs) may reduce the alloy's resistance to stress corrosion.

The most widely used temper of 2219 in LV tankage is T87. In this grade, in-plane A-basis ultimate tensile strengths are 63-64 ksi, with B-basis strengths only about 1 ksi lower, indicating very good control of strength variability. Yield strengths are around 51 ksi. Elongation to break is 6-7% for the thinner gauges of plate. As with all aluminum alloys, the elastic modulus is around 10.5×10^6 psi, one-third that of steel, so significant springback often occurs in cold-formed parts. Very thick Alloy 2219 shapes have lower yield and ultimate strengths than thinner ones. Thickness at the time of solution heat treatment, not the final machined thickness, should be taken into account when establishing design allowables.

The tensile strength of aluminum alloys is increased by cryogenic temperatures. For example, at LOX temperature (-297°F), 2219-T87's ultimate and yield strengths are 20% higher than at room temperature. At LH_2 temperature (-423°F), the strengths are more than 30% higher. This increase in strength is frequently taken credit for in design margin calculations. However, large, thin-walled tanks may buckle at a lower compressive load than that necessary to cause a failure in strength. In such cases, it is the cryogenic elastic modulus, not the cryogenic strength, that determines the compression capability of the tank. The increase in modulus is not as impressive as the increase in strength; for 2219-T87, it is only about 10% at LH_2 temperature [8].

One problem associated with the use of Alloy 2219 has been the difficulty of chem-milling in the T3 temper. This problem was encountered with the hydraulic bulge-formed and chem-milled dome gores of the S-IC, and ultimately led the designers of the Shuttle External Tank to abandon chem milling and adopt the more capital-intensive, but easier to control, stretch forming process [51]. Alloy 2219 is also subject to

7. Materials for Launch Vehicle Structures

surface corrosion, especially in the clad grades. It was found that foam insulation on a 2219 substrate resulted in collection of a chloride-rich liquid in the salt air environment of the southern United States which caused extensive corrosion after exposure of many months [51].

The other workhorse aluminum alloy for stable tank designs is 2014. Alloy 2014 has copper as a principal alloying element (4.4%) but at a lower level than 2219 (6.3%) [52]. It was developed in 1928 primarily for use in aircraft structures as forgings and extrusions; for LV tanks, the sheet or plate forms are used. Alloy 2014 generally has higher strength than 2219: in the T6 temper, its A-basis tensile strength is 64-67 ksi, a few percent stronger than 2219 [8].

Alloy 2014 is a precipitation-hardening alloy. Unlike the widely used 2219-T87 grade, commercial tempers of 2014 are not cold-worked. As with 2219, considerable springback may occur after cold forming, and this is typically corrected by “overforming” [52]. Both 2219 and 2014 are easily machinable, which is important in designs with integrally machined stiffeners.

Alloy 2014 has been used in the Titan II booster, the Saturn S-II stage, and the Saturn S-IVB stage [49]. The Saturn I, designed in the late 1950s, used the Al-Mg alloys 5456 and 5083, but these are rarely considered any more due to their lower cold notch toughness, and greater susceptibility to corrosion. However, they are more weldable than the 2000-series alloys. That is, they lose proportionately less strength and ductility in the welded condition [49]. Both 5456 and 2014 appear to have been early candidates for the S-IC stage [53], but 2219 was ultimately selected. Another aluminum alloy, 6061, was used on the Agena tanks [54]; while this alloy still has some applications in other vehicle structures, it is no longer used for tanks.

Welding processes for tanks have an influence on materials selection. Historically, most tanks have been fusion-welded. The S-IC stage used gas tungsten-arc welding (GTAW) to join 2219 panels [51], a practice that continues to be popular. More recently, plasma arc welding has been implemented. Variable-polarity plasma arc (VPPA) welding, in which the arc polarity is periodically changed to reduce the accumulation of dross, was successfully implemented on the Shuttle ET and Delta IV programs and has also been used to join Al-Li alloys [55]. The large Soviet/Russian Energia booster used electron-beam welding in its tanks [2].

Within the last ten years the development of friction stir welding (FSW) has been a major advance in tank manufacturing. FSW was developed in the 1990s and is now used in production on several LVs. In this process, a rapidly rotating pin moves along the weld lands, mixing clean base metal, which welds spontaneously. It produces a higher-strength and higher-ductility joint than fusion welding because the material is never melted [56]. FSW is particularly attractive for aluminum alloys because of their low hardness. FSW was introduced into production on the Delta II program in 1997 [55]. But FSW is more sensitive to weld land alignment deviations than fusion welding.

Aside from 2000-series aluminum alloys, the material with the widest current application to propellant tanks is the aluminum-lithium (Al-Li) series of alloys. These alloys contain only a small amount of Li by weight (about 1%), less than their Cu content of 2-4%, but they are known as Al-Li alloys to contrast them with non Li-containing alloys. An Al-Li alloy was developed specifically for aerospace applications as early as the 1950s, but problems with fatigue, fracture and weldability precluded its widespread use in the United States until the 1990s [51]. While all wrought alloys are anisotropic in strength and stiffness to some degree, Al-Li is anisotropic enough that it must be structurally analyzed as such. One study found that 2195 Al-Li extrusions [57] had direct and off-axis strengths differing by as much as 20% depending on the depth through the section.

In the early 1990s, funding became available for a major redesign of the Shuttle ET with the primary goal of reducing weight. Weight reduction became necessary when it was decided that the ISS would be put into a high-inclination orbit accessible to Russian launchers; the Shuttle then had to reduce its empty weight to be able to reach the ISS. A series of weldable Al-Li alloys under the Weldalite trade name was available to Lockheed Martin, prime contractor for the ET. The redesigned tank was given the abbreviation SLWT, for super-lightweight tank.

The Al-Li alloy 2195 ultimately selected for parts of the SLWT is lighter than the formerly used Alloy 2219, but has yield strength about 20% higher at both ambient and cryogenic temperatures [12]. It is also about 8% stiffer than 2219. However, 2195 is less formable in the T3 condition than 2219, so an early attempt to simply drop it in as a replacement for 2219 resulted in damaged forming equipment. The

7. Materials for Launch Vehicle Structures

remedy was to solution treat and quench the 2195 into the T0 condition, then stretch form and shape to the T3 condition, and finally age to the T8 condition [51]. Alloy 2195 is also less ductile than the 2000-series aluminum alloys. Ultimately, all of the ET tank barrels as well as the intertank thrust panels were changed to Al-Li.

It was also found that fusion welds on Al-Li were more susceptible to hot cracking than on 2219, and that the subsequent repairs were more difficult. Process changes involving a smaller heat load, a backside inert gas purge, and weld bead planishing were necessary to enable the needed repairs [51]. However, weld quality concerns led Marshall Space Flight Center to investigation FSW for the Al-Li tank components. FSW was implemented on the ET starting in 2002.

Other applications of Al-Li have been the DC-XA and X-33 research vehicles. In both cases, composites were used for the LH₂ tanks, but Al-Li was used for the LOX tanks. Composite LOX tanks require a protective liner, typically a halogenated polymer, to reduce the chance of ignition [58]. The DC-XA LOX Tank was built in Russia from Al-Li alloy 1460 [59].

The Ares I upper stage was a structurally stable, common-bulkhead propellant tank design with friction stir-welded Al-Li 2195 tank barrels and domes. The common bulkhead was a sandwich construction consisting of 2014 facesheets enclosing a phenolic honeycomb core. The bulkhead was to be joined to the barrels by a 2219 Y-ring [60].

In a pump-fed stage, the propellant is held under low pressure in the tanks, then pumped to the injection pressure after it has left the tank. The tanks therefore may be constructed lightly, and stresses due to external flight loading are comparable to those due to internal pressure. In contrast, pressure-fed stages do not have pumps; the propellants are forced into the engine by holding them under high pressure in the tanks. This type of design is used when simplicity and reliability are paramount. Injection pressures for pump-fed engines may be several thousand psi, which would require inordinately heavy tankage. But pressure-fed systems are designed to require only moderate injection pressures. While this is higher than the tank pressure in a pump-fed stage, it is low enough that the tank can be flight-worthy at an acceptable weight. Stresses in tanks for pressure-fed stages are dominated by internal pressure loads.

Some pressure-fed designs have used internal bladders to expel pro-

pellant from the tank rather than externally supplied gas. Many basic design and materials selection aspects are discussed in [54]. Pope and Penner [61] described testing of multilayered bladder materials consisting of various arrangements of polyethylene terephthalate (PET) film, composite balloon film, aramid film and polyimide film. They found through subscale testing that a PET-balloon film fabric provided good performance under cryogenic conditions, with the lowest permeability. Gleich and L'Hommedieu [62] performed similar studies on wire-reinforced metallic bladders of annealed austenitic stainless steel.

Calabro *et al.* [63], in the course of system studies for an advanced pressure-fed cryogenic upper stage, proposed combining a 2219 aluminum LOx tank with a filament-wound graphite-epoxy LH₂ tank in a common-bulkhead design. The LH₂ tank used an internal aluminum foil liner. The working pressure was 270 psi. Thermal insulation was provided by externally applied polyurethane foam.

Many LVs use hydrogen as a propellant in the booster, the upper stages, or both, so the compatibility of materials with hydrogen must be thoroughly understood. Cataldo [64] summarized the findings of several research programs investigating hydrogen embrittlement in high-pressure storage tanks, fasteners, and weldments. Although the focus was on titanium alloys and Inconel 718, useful information is provided on a wide variety of aerospace metals. High pressure was not always a necessary condition for problems with hydrogen compatibility. Hydrogen embrittlement of metallic materials is discussed in Chapter 2.

Balloon Tanks

The Atlas vehicle designed by K.E. Bossart at Convair Division of General Dynamics in the early 1950s is exemplary of this type of design. The other notable application is the Centaur upper stage, also developed by General Dynamics. The Atlas maintained the balloon tank design through several ICBM variants, the early Atlas E and F space launch vehicles, and the Atlas I, II and III commercial space launchers. The Centaur stage still uses the balloon tank design. Balloon tanks require either mechanical tension (“stretch”) or internal pressure to keep them from collapsing under their own weight prior to operation. In operation, the pressure required for propellant feed is sufficient to keep the tank stable under flight loads. The following information is taken primarily

7. Materials for Launch Vehicle Structures

from the review by Martin [65].



Figure 7.4: The Atlas launch vehicle carrying John Glenn to orbit. The balloon propellant tanks can be seen; the LOX tank is forward and covered with frost, while the fuel tank is aft and its shiny stainless steel skin is clearly visible. Public-domain photo by NASA.

Balloon tanks have very thin walls (as thin as 0.01 inch, thinner than three sheets of copier paper) and are built from corrosion-resistant steel. In the Atlas and Centaur, most of the tank skins are made from stainless steel Alloy 301 in the extra full-hard (EFH) grade. Skins that must be formed into a shape other than a circular cylinder, such as conical transitions or domes, are made from 1/2 and 3/4 hard grades, for improved formability. Because the tank walls are so thin, machined reinforcing rings must be placed at locations where external hardware such as feedlines, electrical tunnels, or strap-on booster rockets must be attached. These rings are made from 321 stainless steel, because it is

more machinable than 301. Both 301 and 321 are austenitic stainless steels, whose primary alloying elements are chromium and nickel.

In the very early phase of ICBM development, a vehicle was designed using the balloon tank concept but with aluminum instead of steel as the material. However, comparing the specific strength of 2219 aluminum and 301 EFH stainless steel at LOX temperature,

Alloy	A-basis Yield (ksi)	Density (lb/in ³)	Specific Strength (ksi/(lb/in ³))
2219-T87	60	0.103	583
301 EFH	200	0.286	699

it can be seen that 301 stainless steel offers an advantage, especially in the EFH condition and at cryogenic temperatures. Also, aerodynamic heating of the skin must be considered. The Atlas missile was designed as an ICBM and had to be able to withstand a depressed trajectory that resulted in skin temperatures as high as 700 °F. At this temperature, the stainless steel loses only 17% of its room-temperature strength, while the aluminum loses more than 80% of its strength. Aluminum could only be used if it were highly insulated, at an inert mass penalty.

The 10-foot diameter Atlas balloon tank barrels were constructed from stubby bands 32 inches high. The bands were “stovepiped” together (i.e. inserted into one another a short distance), resistance seam-welded, and then spot-welded on both sides of the seam weld for added strength. The longitudinal welds in the bands and dome gores were resistance butt-welded, and then a doubler was applied with several rows of spot-welds. No filler material was used in the resistance welds, although it was found that placing nickel foil between the workpieces produced stronger spot-welds.

Composite Tanks

While light weight is always a major goal in the design of aerospace structures, it is especially important in launch vehicle stages that ultimately will be propelled to orbit. In staged vehicles, the inert weight of boosters is jettisoned once the booster’s fuel supply is exhausted. However, the orbital stage is not jettisoned, so there is a very high motivation to keep

7. Materials for Launch Vehicle Structures

its inert mass to the absolute minimum. Every pound of inert mass on an orbited stage is one less pound of payload that can be carried.

The vision of a reusable, single-stage-to-orbit (SSTO) vehicle with airliner-like operations has existed since the earliest speculations about space travel. Such a vehicle would have no jettisonable boosters, with all of its inert mass propelled into orbit, re-entering the atmosphere, and returning to Earth to land. Therefore, structural mass efficiency is paramount. Barring unforeseen developments in propulsion technology, any SSTO vehicle must have a structure that is at the absolute maximum efficiency possible with known materials.

The imperative to minimize inert mass has been one of the major reasons so much research effort has been directed toward composites, the other major reason being the ability to fabricate complex cross-sectional shapes with inexpensive tooling and processes. The tensile strength-to-weight ratio of graphite fibers is many times that of the aluminum alloys and steels typically used in propellant tanks.

But the raw tensile strength-to-weight value that is so favorable for graphite fibers can be misleading. To produce a useful structure, the fibers must be incorporated into a matrix; this decreases the tensile strength by about 50% and adds the weight of the matrix, which carries little load. Also, unlike a true pressure vessel, the skin of a pressurized structure will not always be in tension. Compression loading raises the possibility of buckling. While composites with elastic moduli several times that of an equivalent-weight metal design may be produced, it is difficult to control the geometric imperfections that are so damaging to buckling resistance.

The polymeric matrix of conventional composites places an upper limit on the service temperature. Conventional graphite-epoxy composites lose strength and stiffness rapidly when temperatures reach 200 °F to 300 °F, due to softening of the matrix. Thus, composite tanks must be insulated or protected from skin heating by trajectory limitations. This is especially constraining to the design when the trajectory includes re-entry, as it does for a reusable vehicle. Improvements in both thermoplastic and thermoset matrix materials are potentially a means of raising the temperature limit.

Also, especially for tanks of complex shape, reinforced joints are necessary. The need to reinforce these joints and to insulate a composite tank against aerodynamic heating tends to erode the weight advantage over a metal tank. It has been stated that a composite tank can represent

a 20-40% weight saving compared to an equivalent metal tank [66, 67]. In the specific case of the DC-XA, NASA claimed in a press release that the composite LH₂ tank was 37% lighter than the metal tank used on its predecessor, the DC-X [68].

The vast majority of the composite experience base has been with laminates; that is, panels built up from several layers of material manufactured in a previous process. The challenges of joining laminated panels, and their poor interlaminar strength, has led to an interest in braided, woven and knitted textile preforms manufactured by resin transfer molding (RTM) and resin film infusion (RFI) [69]. These preforms offer a way to join laminated panels without a subsequent bonding process or discrete fasteners. They can also be used to fabricate braces and bulkheads that are not panel-like in geometry.

As stated in the previous section, composites were first applied to LH₂ tankage. Since that time, composite tanks compatible with LOX have been developed, but a protective liner separating the composite walls from the LOX was necessary to reduce the potential for ignition [58]. LOX may also chemically degrade the matrix, through oxidation.

The National Aero-Space Plane (NASP) or X-30, a SSTO system contemplated in the 1980s, sought to use composite liquid hydrogen tanks. Hartunian [70] recounted something that often occurs in high-risk developments: despite plans laid by knowledgeable people, significant technical challenges do not come to light until some work is actually done. In the case of the NASP tank, scaling the concept up from laboratory scale to production scale introduced some difficulties. The IM7/PEEK composite initially identified as the one with the best resistance to microcracking could not be scaled up. The cure temperature and pressure, and the required cooling rates, could be achieved at small scale but not at production scale. After the failure to cure the production-scale tank, the engineers changed the PEEK matrix to 8551-7A epoxy. The epoxy matrix design was successfully fill/drain cycled, but the program was canceled for other reasons.

Two more recent programs intended to advance the state of the SSTO art were the DC-XA and the X-33 suborbital technology demonstrators. These programs used composite cryogenic propellant tanks. The DC-XA vehicle flew twice, with the composite tank performing satisfactorily [71], while the X-33 never flew, largely due to development difficulties with its

7. Materials for Launch Vehicle Structures

composite LH₂ propellant tank, including a major test failure.

Most of the interest in composites for propellant tank applications has centered around graphite-epoxy. Both the DC-XA and the X-33 used graphite-epoxy tanks, and the DC-XA also used a composite LH₂ feedline. In addition, the DC-XA used a composite intertank structure. The composite structures on the DC-XA were developed with the aid of rapid prototyping methods [69].

The X-33 tank was a sandwich design with graphite-epoxy facesheets and an aramid-reinforced phenolic honeycomb core. The core contained empty spaces that were not vented. The X-33's development difficulties and 1999 test failure have strongly influenced research in the field since that time. An overview of that design and failure is now presented as a way to introduce the key materials and structures issues involved in composite cryogenic tanks.

Aerodynamics forced the X-33 tank to be structurally much more complex than typical LV tankage. It consisted of a lobed outer barrel constructed from composite sandwich, and monolithic composite internal stiffening frames (Figure 7.5). In addition, bulkheads and thrust tubes were attached to support primary structural, landing gear and control surface loads. The X-33 tank could almost be considered a composite fuselage filled with LH₂.

The X-33 tank was in the process of being qualified in a protoflight program. This entails testing the actual flight article to load levels higher than the maximum expected flight loads, but not as high as a single dedicated test article would be subjected to. The tank had been cryogenically cycled three times, subjected to proof pressure while filled with LH₂ and then subjected to one external test load case while filled. A few minutes after the tank had been drained, it suffered a catastrophic delamination.

It was found that cold gaseous hydrogen had entered the sandwich core from the inner volume of the tank by permeating the inner facesheet. At the same time, ambient nitrogen gas was drawn into the core through the outer facesheet. The permeation processes were abetted due to the strain induced by the test pressure and loads, combined with the low temperatures, which caused leak paths to develop. As the tank cooled to LH₂ temperature as it was filled, the trapped gases condensed into liquids, creating a partial vacuum that drew additional gases into the core. Upon



Figure 7.5: The X-33 liquid hydrogen tank on a test stand at NASA Plum Brook Station, Sandusky, Ohio. Note the complex, lobed shape of the tank. Public-domain photo by NASA.

draining, the tank began to warm to room temperature, and the pressure in the core rose as the liquefied hydrogen and nitrogen warmed up and began to evaporate. The pressure resulted in a sudden debond of the entire area of the inner facesheet. A pre-existing bondline flaw, in the form of a piece of slippery tape found between the core and facesheet, probably contributed to the failure.

This failure mode is called cryopumping. Generally, in the context of aerospace structures, cryopumping refers to the condensation of gas in a void and the drawing in and condensation of additional gas due to the lowered pressure in the void, followed by the possibly destructive rapid venting of the gas upon reheating. In cellular insulations such as polymeric foam, cryopumping occurs when the insulation is cooled by contact with a tank filled with cryogenic propellant, then heated as the

7. Materials for Launch Vehicle Structures

vehicle ascends, the tank empties, and aerodynamic friction heats the insulation. Liquid air condensed in voids in the foam is vaporized and will blow a hole in the foam if it cannot gradually vent.

Crypumping was a known condition that the X-33 design was supposed to accommodate, and the failed core in fact had a measured cryopumping pressure that was lower than the design value, but local, unobservable peaks in the pressure may have exceeded the bondline capability. Despite ultrasonic NDI, the PTFE tape, as well as other debonded areas, were not detected prior to testing. They were only observed after the test article had failed. The possibility of manufacturing flaws difficult to screen by inspection or proof testing has always been a disadvantage of composites, especially in sandwich constructions.

It is a mistake to conclude that the X-33 failure proves composite tanks can never work, because that particular application was much more demanding than conventional applications. It is known that thermomechanical cycling, which is much more severe on a reusable vehicle like the X-33 than on an expendable, is the primary driver of permeation and leaking. After all, composite filament-wound, monocoque solid rocket motor cases have been successfully used for years, and mechanically they are similar to liquid propellant tanks. However, composites are not as clean a solution as they might appear to be from a naive conception of their raw material properties. In particular, the need to characterize and control permeability without the use of a liner has been the thrust of much recent research in composite tanks.

During and after the X-33 program, several research projects have sought to improve the performance of composites in cryotank applications. Heydenreich [72] described system studies carried on in Europe to establish which tankage applications could most benefit from the use of composites. He pointed out the need for a mechanically strong, yet thermally insulating design, suggesting that a liner would be necessary to prevent permeation. He also recognized the fact that composites do not exhibit plastic behavior, which requires a different design philosophy than for metal tanks.

Sankar *et al.* [66] conducted a multiyear research program aimed at developing improved analytical models of gas permeation through composite panels at cryogenic temperatures and under complex, fluctuating stress states. In particular, they examined the effect of interacting distributions of oriented cracks in the different layers of a laminate. Transverse

microcracking due primarily to thermal stress is known to contribute to permeation. A fracture-mechanics based approach was used to predict crack densities and permeation rates. They additionally performed testing that showed cryogenic cycling caused a degradation in the resistance of panels to permeation due to the opening and propagating of cracks. The testing showed that textile (woven) composites had less permeation than laminated composites after cycling; this was attributed to the lack of propagation of transverse cracks.

Morino *et al.* [67] carried out preliminary tests using a subscale tank with a liner, focusing on the Y-joint at the dome-barrel intersection. They noted the difficulty of maintaining a quality laminate in such locations and aimed their testing at this area. They observed matrix microcracking at low stress levels when the matrix was cold.

Graf *et al.* [73], noting the need for leakproof adhesively bonded joints in cryotank applications, tested a double-lap joint design. They showed that the lack of a peel-ply surface preparation, as well as the use of an adhesive primer, reduced the bond strength. Overall, they found, as in other investigations, that cryogenic temperatures reduced the strength of their components by 50% or more. They showed a size effect; that is, the larger the bonded surface, the lower the supported shear stress. Such effects are usually attributable to the greater likelihood of bondline defects as the bonded area increases.

Miller and Meador [74] found that clay-based layered silicate nanocomposites, dispersed in the epoxy matrix, significantly reduced thermal expansion and gas permeability in the resin both before and after cryogenic cycling. The degree of reduction was directly related to the weight percent of nanocomposite. They also found that, while the nanocomposite matrix led to a laminate with lower flexural strength than plain epoxy resin, the nanocomposite retained its strength after thermal cycling. It appears that after cycling, the nanocomposite laminate had strength comparable to the plain resin laminate. However, these encouraging results did not translate to decreased permeability when the nanocomposite matrix was used in a subscale test bottle.

Pavlick *et al.* [75] investigated the strength of advanced matrix materials. The resins were tested in the form of tensile and fracture samples machined from neat plaques. Tensile strength, modulus, elongation to break, toughness and fracture properties were measured at temperatures

7. Materials for Launch Vehicle Structures

ranging from +320°F to −310°F. It was found that cryogenic temperatures tended to increase the strength and decrease the elongation to break of the matrix materials. Trends in fracture properties were unclear. A candidate liquid crystal polymer matrix material was found to be generally more brittle and less tough than the three other resins, all polyimides.

Black [76] discussed recent advances in research on composite tanks for cryogenic fluids. An unlined composite LOX tank for the since-canceled X-34 reusable vehicle was successfully tested for fill/drain cycling and impact resistance. The ability of composite tanks to incorporate more complex shapes than those of metal tanks has been enhanced by in-situ fiber placement, which can produce thick, curved structures that do not wrinkle during cure, and can eliminate the need for debulking. Another new manufacturing method that eliminates the need for debulking is to lay up a panel by ultrasonically bonding thin layers of prepreg tape. Linerless tanks may be possible if toughened, advanced matrix materials are used. Even composite tanks still must use heavy metal bosses for fluid connections. However, composite bosses manufactured by resin transfer molding (RTM) have been tested.

Solid Rocket Motor Cases

Large solid rocket motor cases are discussed in this chapter because of the significant flight loads (in addition to self-generated internal pressure) they carry. Although they usually are “strapped on” and therefore are not in the primary load path, in one vehicle, Ares I, a motor case did form the bulk of the booster primary structure. Solid motors also provide primary structural support in solid-fueled missiles. Because of their size and rigidity, solid motor cases are attractive locations for the attachment of auxiliary flight systems, and they also must support strap-on booster nose cones and aft fairings. In this respect, they have more in common with liquid main propellant tanks than with the combustion chambers of liquid rocket engines. However, they must withstand pressures that can exceed 1000 psi, far higher than the pressures in propellant tanks.

A solid motor case is composed of a barrel section, a forward dome and closure, and an aft dome with provisions for mounting a nozzle. Smaller motors such as the GEM-40, -46 and -60 strap-ons for the Delta II, III and IV, and the Atlas V solid rocket motor, can be produced as

a single, monolithic unit. Very large motors, including the Titan III and IV strap-ons and the Space Shuttle SRBs, must be manufactured in segments in order to be transportable over the road. Motor cases and segments are permanently loaded with propellant by the manufacturer, and therefore must be handled carefully as they are transported to the launch site, where they are assembled or “stacked.”

A “case-bonded” (as opposed to cartridge-loaded) motor typical of those used for LVs consists of an outer shell, closed forward and aft by domes, and the assembled pressure vessel is lined with an insulating material that both protects the case from the heat of combustion and facilitates the bonding of the propellant to the case. The propellant is then cast directly into the lined case and cures to a rubbery consistency. Neither the propellant nor the insulation provides significant strength or stiffness to the motor as a whole, so they are not discussed further here. Additional details are given in Chapter 11.

The pressurized envelope of a motor case is capped by a forward closure, which usually houses the igniter, and an aft closure that must provide an attachment for the nozzle. Also, forward and aft skirts are usually provided for attachment to other vehicle structures. These are integral with the motor case.

Except for the very largest first-stage boosters, solid rocket motor cases are designed based on the pressure stress plus flight loads amounting to some fraction of the pressure stress. As with main liquid propellant tanks, cyclic loading during proof testing may cause flaws to propagate. But solid rocket motor cases are also subject to pressure oscillations at frequencies up to 1000 Hz during the motor burn [77]. Therefore, nondestructive inspection methods of similar type and significance as those previously discussed for liquid propellant tanks also apply to solid rocket motor cases.

Motor cases are generally constructed of high-strength steels, titanium, or filament-wound graphite-epoxy. Pressure stresses usually preclude the use of aluminum except for very small motors. Metal cases may be built from rolled and welded sheet or by seamless methods such as drawing or spinning. The presence of a welded seam lowers the strength of the nearby material and requires heat treatment and careful inspection. Steels that are commonly used are D6AC, the 18% nickel maraging steels, and 4130 alloy [77]. Steels requiring post-fabrication heat treatment may pose a

7. Materials for Launch Vehicle Structures

problem because of the very large diameter of the finished product. There is a limit to how large the structure can be before exceeding the capacity of commonly available heat treatment facilities.

Solid rocket motor cases were one of the earliest applications of filament-wound composites technology. Peters [78] states that motor cases “were primarily responsible for accelerating filament winding from a laboratory curiosity to the major industry it is today.” As with propellant tanks, a major reason composites are attractive as a material for motor cases is the ability to orient the strong direction of the material along the direction of highest loading. This leads to greater structural efficiency than is possible with an isotropic material. In motor cases, more so than other structures, it can be stated with high confidence that the state of stress is close to biaxial, with the axial stress about half of the hoop stress. Flight loading is small compared to internal pressure and will alter this ratio but little.

The titanium alloys and high-strength steels commonly used for motor cases have specific strengths of about 850 ksi/(lb/in³), whereas composites can achieve 3-5 times this value. Other reasons to use composites include lower-cost and more adaptable tooling, relatively low-cost raw materials, and imperviousness to corrosion. The thermal environment for motor cases is not significantly different from that of non-cryogenic primary structure. Although combustion temperatures are as much as 4000 K, this extreme temperature does not have time to penetrate through the very poorly conducting solid propellant and insulation to the case.

Several programs, including Titan and Space Shuttle, have developed composite filament-wound replacements for motor cases that were initially metal. Not all of these new designs were put into production. In the case of the Space Shuttle, the filament-wound motor offered a definite mass fraction advantage over the existing design, but the extra capability was only needed for polar orbit launches from the Western Range, which were canceled after the *Challenger* failure [79].

The Delta II uses up to nine large strap-on GEM-40 solid rocket motors. The GEM-40, -46 and -60 have graphite-epoxy filament-wound cases. Filament-wound cases have even been able to meet the very stringent mass efficiency requirements of upper stages. The Inertial Upper Stage (IUS) developed as an upper stage for both Titan and Shuttle, incorporated two aramid-epoxy filament wound motors.

Gargiulo *et al.* [80] compared failure envelopes generated by several commonly used composite failure criteria to test data for pressurized filament-wound tubes. Two early studies of materials selection for solid rocket motor cases are [81] and [82].

Pionke and Garland [83] compared D6AC and 18-Ni maraging steel from the standpoint of subcritical crack growth behavior in motor case applications. This research was conducted in the course of early Space Shuttle system studies. They found that D6AC had inferior corrosion and stress corrosion resistance, and also experienced a decrease in cycle life when exposed to temperatures needed during refurbishment operations.

7.6 Feedlines, Small Lines and Pressure Vessels

Many tubes and pipes are necessary to supply fluids to the propulsion and guidance systems. These components range from small tubes less than an inch in diameter to main propellant feedlines, which can be 18 inches or more in diameter. The larger lines frequently must have gimbals or flexible sections so that thermal and mechanical stresses do not build up, especially where the lines connect a strap-on to a main booster that may experience large relative motions. Also, lines may connect to the inlet valve on a gimbaled engine that undergoes large motions.

Feedlines

Feedlines are different from other pipes and tubes due to their large size, higher criticality and high flow rates. Operating pressures are similar to those in the tanks. Some lines are downstream of pumps and the pressure can be several thousand psi, but pipes downstream of feed pumps are usually considered part of the propulsion system and therefore fall outside the scope of this chapter. Either the fuel or the oxidizer tank may be in the forward position. The feedline from the forward tank has a downcomer that may run along the side of the aft tank, or may penetrate the tank. The downcomer can be more than 50 feet long.

Feedlines are usually constructed of 321 corrosion-resistant steel (CRES), although 347 CRES, Inconel 718, Hastelloy and A-286 have also been used [84]. Inconel 718 and Hastelloy are especially suited to areas experiencing fluctuating loads and corrosive environments. Feedlines can experience a high fluctuating load component relative to the mean

7. Materials for Launch Vehicle Structures

load, because of dynamic excitation and flow-induced vibration. They may also vibrate during pogo, which is an undesirable resonant interaction between the motion and pressure of the fluid, and the structural modes of the feedlines or adjacent hardware.

The DC-XA included a composite LH₂ feedline among the technologies it demonstrated [85].

Metals for feedlines must have high ductility because of the need to form elbows and bends. They must be formable, weldable, and compatible with common lubricants. They must also have adequate performance at low temperatures, when cryogenic fluids are involved. They must be chemically compatible with the working fluid. A particular problem is hydrogen embrittlement (see Chapter 2); Inconel 718 is incompatible with high-pressure hydrogen for this reason. A corrosive or chemically active environment can significantly lower the fracture toughness of materials. Also, some fuels undergo rapid or even explosive reactions when they contact certain metals. For example, the breakdown of certain hypergolic propellants is catalyzed by some of the trace alloying elements present in many metals.

Cryogenic lines may require insulation, whether they are inside or outside the vehicle shell. Insulation is required to minimize boiloff, maintain the fluid within the required temperature and pressure, and prevent geysering. Geysering is when the fluid in a vertically oriented line partly vaporizes and the bubbles rise and rapidly exit the top of the line. Insulation on feedlines uses much the same technology as the lightweight thermal protection systems for vehicle primary structure.

Both large feedlines and smaller tubes may be subject to safety factors and testing requirements that are quite different from primary vehicle structure. Lines that are small and can be pressurized when personnel are nearby may be held to safety factors as high as 4.0. When EWR-127 applies, many safety precautions are required. Proof pressure testing is almost always mandatory, and the many system functional and leak checks that are carried out can consume a significant portion of the safe life of a small line.

Pressure Vessels

Launch vehicles need to store small quantities of hydraulic fluid, secondary propulsion or reaction control propellants, helium for system

pressurization, and the like. High pressures may need to be withstood. The classic design for this application is a Ti-6Al-4V welded sphere.

A more mass-efficient design, widely used today, is the composite overwrapped pressure vessel (COPV). This design uses a very thin metal shell only as a leak liner. The membrane strength is provided chiefly by a filament-wound composite layer on top of the metal liner, usually graphite-epoxy or aramid-epoxy. The liners may be titanium alloys or Inconel. The two-layer construction allows the liner to be placed in a state of residual compression, by initially pressurizing the tank beyond the yield point of the liner. This process is called *autofrettage* or *sizing*. When the pressure is removed, the overwrap elastically recovers, imposing a compressive stress on the liner. In subsequent pressure cycles, the liner will not go into tension until the sizing pressure is exceeded. This process greatly improves the pressure and fatigue capability of the liner. Obviously, if autofrettage is to be done, the material selected for the liner must have a stress-strain relation that permits it. Low variability in the yield strength and draw properties is needed in order to keep the results of the autofrettage operation within control.

The inspection and safe life analysis of COPVs have been extensively studied, and specialized standards exist; see, for example, [86, 87, 88]. However, with the liner strongly compressed when the vessel is empty, liner buckling must be prevented. A good, continuous bond of the liner to the overwrap is necessary. Unbonded areas due to inadequate adhesion or protruding weld beads on the liner can cause the liner to buckle. The leak-before-burst requirement is not entirely straightforward to apply to COPVs because of the separate liner and overwrap.

7.7 Unpressurized Structure

Here, unpressurized structure means passively vented structure that experiences low pressure differentials, no more than a few psi. For these structures, pressure is not a driving factor in design. Examples are fairings, nose cones, skirts, adapters, thrust structures, wings and control surfaces. Usually, at launch, a mixture of gases, primarily air, exists at near-atmospheric pressure in the interior spaces of these structures. These gases may be very cold if near a cryogenic tank and may contain gaseous propellants or oxygen due to prelaunch venting operations.. After

7. Materials for Launch Vehicle Structures

launch, as the vehicle ascends through the atmosphere, the internal gases escape the structure through vents or natural leak paths.

Unpressurized structures may need to maintain a controlled interior temperature and humidity environment, as with a payload fairing, or there may be no control at all of the interior environment, as is usually the case with intertanks and thrust sections.

As with all airborne structure, the strength-to-weight ratio is the most important design characteristic, and when liquid propellants need not be contained, there is more freedom to optimize the materials and structure for light weight. Therefore, unpressurized structures have seen greater use of composites, and the stronger grades of aluminum, whose lower fracture toughness is less of a disadvantage than it would be in structure that sees pressure cycling, may be considered. Lighter designs can result.

The 7000-series aluminum alloys are often used in unpressurized structure. These alloys have zinc as their major alloying element, and have a much higher static strength than the 2000-series alloys used in propellant tanks. However, the 7000-series alloys are not as resistant to damage from repeated loading as the 2000-series alloys, and have less favorable cryogenic properties.

Intertanks, Skirts, Adapters, etc.

A space launch vehicle is, functionally, a number of tanks connected in series, with an engine at the aft end and a payload at the forward end. The structures used to connect the primary functional pieces are known variously as intertanks, interstages, engine sections, skirts and adapters. The generic term “adapter” will be used to refer to any of these types of structures.

Adapters may be simple cylindrical shells providing a space for the end dome of a tank, or they may support feedlines, pneumatic and hydraulic lines, wire harnesses, and other items on internal brackets or shelves. Often the umbilical connections that supply the vehicle with ground electrical power and provide propellant fill and drain capabilities are located in adapter structures. Because of the available internal space, guidance and navigation hardware, telemetry equipment, inert gas tanks, and hydraulic pumps are often located in adapters. Thus, an adapter may have an outer shell that is primary structure and inner shelves or brackets that are secondary structure.

Armstrong *et al.* [89] examined the use of a beryllium-aluminum alloy for use in lightweight stiffened cylindrical barrels, particularly from the standpoint of cost. Both integrally machined orthogrid designs and bilayer corrugated-smooth designs were considered. They concluded that the beryllium alloy would be 50% lighter than an equivalent-performance aluminum design, but as discussed earlier, beryllium dust is toxic and the expensive safety measures required in manufacturing tend to cut into its inherent advantages.

Composites are used to a much greater degree where there is no need to contain a liquid. Therefore, they have many applications in unpressurized LV structures. These applications are similar in requirements and performance as the use of composites in aircraft, the pros and cons of which (weight saving, part count reduction, ability to fabricate complex one-piece shapes, etc.) have been addressed in other chapters. Large composite structures pose design, manufacturing and maintenance challenges that are different from those for metals. Vosteen and Hadcock [90] surveyed industry experts and concluded that using composites requires a period of materials development before product development begins, that scale-up to production can be challenging, that bonded and fastened joints require more precision than in metal structures, and that tooling must be adaptable to allow design changes, control dimensions and adjust for springback.

LVs generally experience a greater temperature range than aircraft. Composite structures on an LV may be close to cryogenic propellant tanks; conduction through the structure and cold vapors emitted during fueling can result in extremely low temperatures. During atmospheric flight, an LV proceeds through hypersonic speeds, and without some means of insulation, heating due to aerodynamic friction can raise the temperature of composites well beyond the softening point of the matrix. Therefore, the low- and high-temperature behavior of composites is relatively more important in LVs than in aircraft.

Adhesively bonded joints, as well as adhesive bonds of core materials to composite facesheets, are especially susceptible to strength reduction at extreme temperatures. It is expensive enough to adequately characterize a bonded joint at room temperature, but when large temperature and humidity ranges must be considered, the task becomes that much more involved.

7. Materials for Launch Vehicle Structures

Kobayashi *et al.* [91] discussed the development of a composite interstage for the H-2A vehicle. The interstage shell was a foam-core, graphite-epoxy facesheet sandwich manufactured by co-curing. The role of geometric imperfections in the buckling capability was investigated. A good description of the structural qualification test is given, in which cryogenic temperatures were imposed at the aft end of the interstage to simulate the in-flight conditions due to an adjacent propellant tank. Such approaches are often necessary in structural qualification tests for launch vehicles.

Payload Fairings and Nose Cones

A conical or tapered shell is used to provide a low-drag shape for the forward end of the vehicle and to protect enclosed payloads during ground handling and atmospheric flight. When a payload is enclosed, the structure is known as a payload fairing or shroud; when no payload is inside (as at the forward end of a strap-on booster), it is called a nose cone.

Nose cones are permanently attached to the strap-on booster and go with the jettisoned boosters when they have completed their burn. Payload fairings are jettisoned once the vehicle has ascended out of the atmosphere and air drag has ceased. Since a nose cone does not need to protect a payload, the functional demands placed on it are less stringent. Another application is the nose cone of a vehicle that undergoes a head-first atmospheric re-entry. This type of nose cone must be able to resist the extreme heat and pressure of re-entry, and must be constructed of heavy heat-sink and shielding materials. Therefore, it is a quite different structure from a nose cone that must function only during ascent. A very early study of materials for this type of nose cone is given in [92].

Even during ascent, nose cones are subject to high heat fluxes, and therefore must incorporate heat-resistant materials, especially at the apex. The Space Shuttle Orbiter nose cone is made of reinforced carbon/carbon, which can withstand temperatures exceeding 3000 °F. Carbon/carbon is a fibrous composite consisting of graphite fibers in a pyrolytic graphite matrix. Expendable vehicles may use superalloys or other heat-resistant metals at the nose cap.

Payload fairings, being at the extreme forward end of the vehicle, do not need to sustain as much axial load as other structures. Therefore, stiffness is relatively more important than strength for a fairing. The

fairing must maintain the shape of the payload compartment so that there is no danger of contact or interference between the payload and the fairing. It must be able to resist the very high-intensity sound waves (160 dB or higher) that reverberate around the launch pad after engine ignition but before the vehicle has risen above the surrounding terrain. These sound waves can be intense enough to excite panel vibrations on the fairing. The fairing may be required to attenuate the liftoff acoustics to protect the payload. The fairing must also be stiff enough so that it does not grossly deform during jettison; the motions and deformations should be linear and easily predictable.

A payload fairing design used on the first Atlas-Centaur launches was made of fiberglass [93]. However, increasingly stringent payload protection requirements and the need to reduce weight whenever possible led to the use of more advanced materials, in sandwich or stiffened shell designs as a rule. The core and facesheets of sandwich shells are often composed of different materials, such as laminated composite facesheets over a phenolic or aluminum honeycomb core, or a foam core. Such constructions require the joining of dissimilar materials, usually by adhesive bonding or co-curing, and may suffer from corrosion or stresses induced by differential thermal expansion. These problems may be solved by using the same material for both the core and facesheets.

The Ariane 4 fairing, a conventional design that is 20 years old but can still be considered state-of-the-art, is described in [94] in the context of a separation test. The fairing shell is largely made of a sandwich of graphite-epoxy facesheets with an aluminum honeycomb core. The forward end of the fairing is an aluminum skin-stringer design that has a layer of cork insulation. This is a less expensive and possibly lighter approach than using a superalloy nose cap. The fairing-vehicle separation system consists of tension belt or clampband that secures the aft end of the fairing to the rest of the vehicle under tension provided by two steel bolts.

The Russian Soyuz LV has payload fairings whose shells are sandwich structures composed of an aluminum skin with aluminum honeycomb core [95]. Schwingel *et al* have described an experimental structure composed of an aluminum foam core with aluminum facesheets [96]. The sandwich layup was manufactured by rolling the facesheets over a layer of mixed aluminum and gas-generating material. In a subsequent foaming process,

7. Materials for Launch Vehicle Structures

the sandwich was heated until the gas-generating material was activated, causing bubbles to expand in the core and increase the thickness of the sheet without an increase in mass. The large foam cells produced by this process were about as big a honeycomb cell. A prototype conical adapter was built using this process and successfully tested to about half of the limit loads applicable to the conventional structure it was meant to replace. Homogeneous core/facesheet sandwich structures such as these overcome the problems of material incompatibility, but cannot be tailored as precisely as sandwiches with differing core and facesheet materials.

Lane *et al.* [97] investigated a fairing design composed of tubes joined into a sheet, subsequently formed into a cylindrical barrel. The tubes were then punctured on the inside of the barrel to reduce the acoustic levels inside the barrel. This design, known as the chamber core fairing, is intended to provide acceptably low sound levels inside the fairing without the need for the usual nonstructural acoustic blankets. They built a laboratory-scale specimen and measured noise reduction equal to that provided by blankets for low-frequency noise. The specimen was constructed of inner and outer filament-wound facesheets with composite tubes between them. There may be difficulties in integrating the cylindrical chamber-core barrel with the required conical shape at the forward end of the nose.

Ochinero *et al.* [98] described the design optimization and subscale wind tunnel testing of an unconventional Large Asymmetric Payload Fairing intended to accommodate very bulky payloads.⁶ They discussed an optimization procedure that resulted in the selection of carbon fiber reinforced facesheets and a Rohacell foam core. This design was governed strongly by buckling rather than strength, which is typical for payload fairings. Consideration was given to buckling behavior beyond the elastic stability limit (postbuckling), which has been applied in practice to balloon propellant tanks but is not usual for other structures.

The use of a Rohacell foam core highlights important considerations, discussed in more detail in the following section, related to core materials. A primary reason for using Rohacell for this application was the relatively low knock-down factor imposed by the program. Program requirements

⁶Material on the Large Asymmetric Payload Fairing and the subsequent section on core materials and inserts were contributed by Tomoya Ochinero and Eric Ruolo, Structural Mechanics Corporation.

dictated the use of core material-specific knock-down factors on the strength of the sandwich panel to account for separation between the core and the facesheets due to such variables as manufacturing imperfections, microbuckling, and moisture entrapment. A more traditional honeycomb core has a tendency to entrap moisture in the cells of the core material. The entrapped moisture can evaporate as the payload fairing is subjected to the high temperatures and low pressures of ascent. Without adequate venting features to relieve the subsequent pressure rise within the core materials, the facesheets can become separated from the core material. For this particular application, the program dictated a significantly larger knock-down factor for an aluminum honeycomb core than for a Rohacell core. It is academically interesting to note that the fairing would have been lighter if it had been designed using aluminum honeycomb core if only the knock-down factors were equal.

Another interesting note that highlights the struggle between idealized design optimization and the realities of manufacturing and operational requirements on this application is the uniform thickness of the core. The optimization analysis showed that a significant weight savings was achievable by tailoring the core thickness to vary with respect to location on the fairing. With Rohacell, it is easier to continuously vary the thickness of the core than with an aluminum honeycomb core. However, the manufacturing constraints on this program required a uniform thickness continuous core. This resulted in a compromise where the core is thicker in many regions where a thinner core would have sufficed. It is notable that despite these design constraints, a fairing with twice the volume of the standard fairing was achieved with only a 33% weight increase.

Core Materials

Core material is used to separate thin composite facesheets and increase the structural efficiency in bending. The purpose of this core material is to tie the facesheets together in shear, thus allowing them to work together in bending. For this reason, when modeling, the properties of the core must be properly taken into consideration. One often overlooked core property is the in-plane modulus of aluminum honeycomb cores. Facesheet-stabilized aluminum honeycomb has a significant in-plane modulus that must be accounted for when conducting thermal analysis or thermal distortion analysis of sandwich parts with thin facesheets. A

7. Materials for Launch Vehicle Structures

good reference to estimate the modulus in the absence of test data is [99].

Incomplete bonding between core edges and facesheets is often the source of many manufacturing induced flaws which cause disbonds and panel failures. These can be mitigated by using reticulation to premelt one layer of film adhesive on the bare core. This increases the bond fillet between the honeycomb and facesheet and has been shown to dramatically increase sandwich panel integrity. The downside is the increase in processing time and the use of twice the number of film adhesive layers, which increases the mass of the panel and adds more high-CTE adhesive into the panel.

For large panels and complex sections, core splices are required. The need to use separate core sections and then bond them together with foaming adhesive adds another design detail with challenging analysis requirements. For most aerospace applications, the foaming adhesive has stronger shear strength than the core, so if the dimensions of the splices are controlled to ensure proper adhesion, the core splice is stronger than the base materials. Splices should be designed to be away from any load introduction points and as far away from highly loaded regions of the panel as practically possible.

With sandwich structures that ascend to outer space or have rapid depressurization requirements, vented core is required to prevent the facesheets from blowing off. An approach to compute this failure mode is presented in [100]. The vapor needs to have a pathway to ambient, requiring edge closeouts to also be vented. Mylar closeout tapes come perforated for such applications.

The core out-of-plane shear strength is utilized to introduce out-of-plane loads via potted inserts. Potted inserts are placed in sandwich panels to connect ancillary components such as equipment boxes to the panel and provide load paths for panel to panel connections. Most companies have proprietary insert designs, but off-the-shelf designs are sold commercially. The analysis of these joints is complicated and is described in great detail in [101]. Test data for these joints is required to validate the design before production. Attention should be paid to the potting compound for this style of insert. With extreme thermal environments, the out-of-plane CTE difference can cause the potting compound to either shear the core or force the failure of the core-to-facesheet bond. Potting compound weight can also become a major driver

to otherwise highly efficient sandwich panel construction, as hundreds or thousands of inserts may be added to a lightweight panel to provide fixation to all the equipment that must be placed on it.

With the proper attention to the additional design and analysis details of sandwich panel construction, weight savings can be realized over traditional structures. The designer must be vigilant to ensure that the additional failure modes and behavior of the structure is well understood to prevent failures.

7.8 Thermal Protection and Insulation

Thermal environments are a significant factor in materials selection for LVs. Most liquid-fueled vehicles use cryogenic LOX at -297°F as the oxidizer, and some use LH_2 fuel, at -423°F . Even though cryogenics are loaded only a few hours before launch, there is ample time for the tank walls, domes, and adjacent hardware to become extremely cold. Venting and leakage of boiled-off propellants create plumes of cold gas that may surround vehicle structures and cause cooling of areas not in direct contact with liquid propellants. Insulation, typically in the form of closed-cell polymer foams sprayed on or bonded on as pre-cured panels, is used to protect hardware from extreme cold and to manage the boil-off of loaded propellants before and during launch.

All LVs must ascend through the atmosphere, typically for two minutes or so. The competing effects of decreasing air temperature with altitude, and increasing frictional heating with acceleration, can cause structural skin temperatures to decrease or increase. Insulation serves to moderate the temperature of the structure during this period. Thus, the insulation applied to a cryogenic propellant tank needs to retain acceptable mechanical and thermal properties at temperatures ranging from as low as -423°F to plus several hundred degrees F.

For the two commonest structural materials, aluminum and graphite-epoxy, temperatures must be kept below about 200°F in order for the structure to retain sufficient strength and stiffness. Aluminum is more tolerant of heating than graphite-epoxy, but still weakens appreciably when temperatures exceed 200°F . High-strength steel is less affected by high temperatures. In some areas, such as the forward end of a nose cone, or an area subject to a standing shock wave, temperatures can be high

7. Materials for Launch Vehicle Structures

enough to require high-temperature (refractory) alloys or carbon/carbon. The leading edges of the Space Shuttle wings, and its nose cap, are made of carbon/carbon with a silicon carbide coating to prevent oxidation. The nose caps of expendable vehicles may be made of beryllium alloys or high-temperature superalloys. However, exposure to high temperatures is brief, so time at temperature is usually not a consideration except after many flights of a reusable vehicle.

The most widely used material for expendable LV thermal protection systems (TPSs) is polyurethane foam. These foams can be sprayed on, poured into molds placed over vehicle features, or bonded on in the form of pre-cured sheets. Foams suitable for use in TPS applications are relatively rigid. Their microstructure consists of packed bubbles or closed cells with polyurethane walls. The polyurethane itself is created by the catalyzed reaction of a polyisocyanate with a polyol. During the casting process, which takes place either directly on the structure when foam is sprayed on or poured in place, or in a factory where pre-cured sheets are made, two parts are mixed. One part is the polyisocyanate, and the other part is the polyol, catalyst, blowing agent, and surfactant. The cells are generated when the blowing agent, suspended in the liquid mixture, expands. When the mixture cools, rigid-walled cells remain, initially containing mainly the blowing agent. As time passes, the blowing agent gradually diffuses out of the cells, and air diffuses in. By the time the foam is put into service as an insulator, the cells may still contain mostly blowing agent, or a mixture of blowing agent and air. The insulating characteristics of the foam can thus change with time, because the thermal properties of the changing cell contents play a significant role.

Until the early 1990s, the most common blowing agents in TPS foams were the chlorofluorocarbon (CFC) refrigerants CFC-11 and CFC-12. These agents, while non-flammable and non-toxic, were recognized as damaging to the ozone layer and were gradually banned in some countries.⁷ Manufacturers no longer able to obtain CFCs sought substitutes such as hydrochlorofluorocarbons such as HCFC-141b, but these, too, were eventually banned. Changes to the blowing agent require the foam to

⁷With regard to the major LV manufacturing countries, the United States and France banned CFCs by 1996. Russia and Ukraine were attempting to eliminate the substances but having some difficulties achieving full compliance. CFCs were still available to Chinese manufacturers as of this writing [102].

be requalified for its intended use. Different blowing agents generate different cell sizes and shapes, and affect the thermal properties of the insulation. Requalification tests may indicate that process changes are needed to maintain the foam's performance.

By varying processing parameters such as flow rate, temperature and ambient curing conditions, a variety of foams can be generated. A surfactant may be used to control the size of the cells. The stiffness of the cell walls themselves is a function of the precursor compounds and the ratio and conditions under which they are mixed. The stiffness and strength of the foam is a strong function of the cell size: smaller cells mean a denser, stiffer and stronger foam.

Over smooth, featureless areas, sprayed-on or bonded-on foams are usually used. Sprayed-on foam is applied in several passes; in the time between passes, the exposed surface of the previous pass can partially cure. A "knit-line" may then form at the boundary surface between two passes, consisting of two adjacent layers of aligned cell walls that appear as a thickened solid wall running through a field of randomly oriented cells. As the foam rises, the forces of gravity, surface tension and internal pressure create cells of dispersed size that tend to be oblong, with the long axes aligned in what is called the rise direction. Noever *et al.* [103] showed microphotographic studies of the effect of gravity on the cell size, shape, and void frequency of foams. Their control sample was manufactured in zero gravity during a sounding rocket flight.

The existence of a distinct rise direction has to do with the fact that the liquid foam has to be constrained into the desired shape, by the structural surface, a partially cured previous pass, and/or a mold. The rise direction and the knit-lines result in anisotropic mechanical and thermal properties.

When complex shapes such as flanges or fastener heads must be insulated, foam is usually poured into molds so that it can closely conform to the underlying surface. Whether foam is poured or sprayed, when the structure has a complicated shape, voids may occur due to the inability of the foam to conform exactly to the surface. Voids may also occur between spray passes and simply as enlarged cells, which will develop in scattered locations due to the slightly incomplete mixing of components. Also, knit-lines will exist wherever a poured area meets a separately poured area, or a sprayed area. Machining or shaving may be necessary

7. Materials for Launch Vehicle Structures

to achieve a low-drag outer profile for both poured and sprayed foams.

The various failure modes of foam insulation received intense study following the *Columbia* accident. Stresses sufficient to fail foam may be caused by cryopumping, thermal expansion, flexing and stretching of the structural substrate, thermal cycling, pre-existing flaws, voids and unbonds, and probably several other failure modes that have yet to be conceived. Bednarczyk *et al.* [104] provided a discussion of the failure modes from the micromechanics viewpoint along with an analytical framework for predicting failure under complex combinations of stress, temperature and pressure histories.

The Space Shuttle contains both major types of TPS: a low-strength, lightweight layer of foam on the expendable External Tank, and more capable, reusable insulation on the Orbiter. The Orbiter is not only reused, it also must withstand the rigors of atmospheric re-entry, which are a far more challenging thermal environment than launch. Figure 7.6 illustrates the location of the different types of TPS on the Space Shuttle Orbiter.

Re-entry TPS technology for reusable launch vehicles has its roots in the (primarily ablative) TPS designs for the early expendable capsules. A summary of the state of the art in ablative heat shield materials for re-entry vehicles was given by Bauer and Kummer [105]. They described the design of a low-density, filled silicone ablative material cast into a nonmetallic honeycomb reinforcement, bonded to a plastic sandwich structure, as applied to the Gemini spacecraft. This was an advance over the Mercury heat shield, which was a glass-phenolic, and a step in the direction of the Apollo Command Module heat shield, which was silica fiber-epoxy resin again cast into a non-metallic honeycomb support structure. These early ablative systems were extremely heavy. The Apollo shield made up almost a third of the total weight of the Command Module.

A reusable TPS with a great deal of operational experience is the ceramic tiles covering most of the Space Shuttle orbiter. The development of these tiles was a major pacing item in the Shuttle program as a whole. There are actually four different types of tiles, with differing capabilities, used in different areas. All of the tiles are composed of amorphous silica fibers with a 0.015-in-thick reaction-cured borosilicate glass coating on the side facing the atmosphere. The system is tiled, rather than a

7.8. Thermal Protection and Insulation

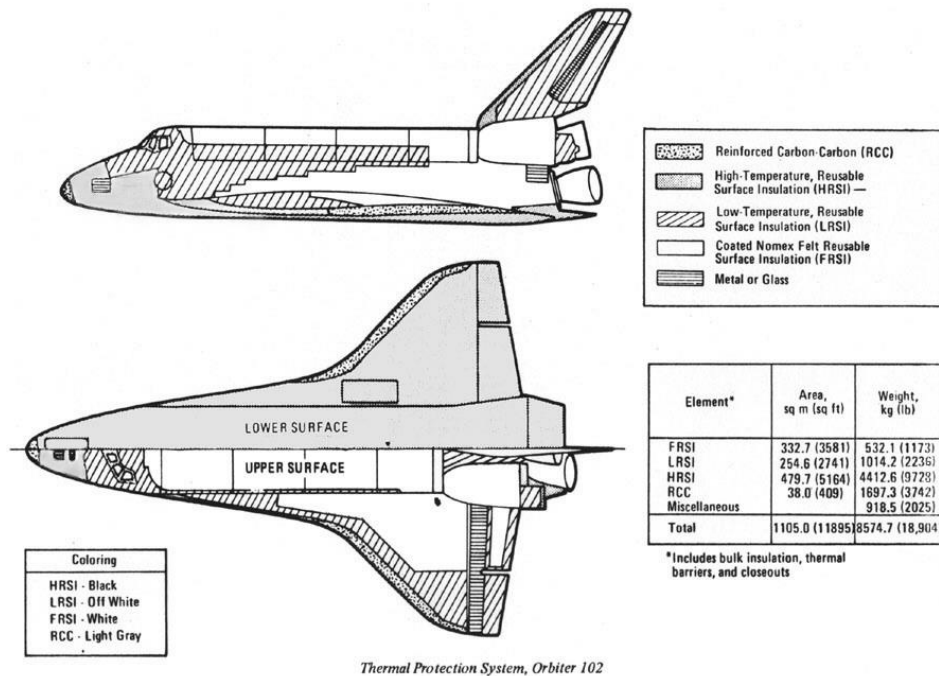


Figure 7.6: Thermal protection system of the Space Shuttle Orbiter. Public-domain graphic by NASA.

continuous sheet, in order to allow thermal expansion of the substrate and individual replacement of damaged tiles. In low-temperature areas (750 °F to 1200 °F), the tiles are colored white, whereas in high-temperature areas (up to 2300 °F) the tiles are colored black to improve radiative heat transfer [106]. The rest of the Orbiter acreage is insulated with flexible blankets.

Carbon/carbon can endure higher temperatures than any other aerospace structural material, up to 3000 °F. It is relatively expensive, difficult to work with and subject to oxidation. Titanium and the nickel superalloys are the next most expensive. Being metals, they are strong and can be worked with conventional tooling, but they are also heavy. More advanced concepts have involved non-metallic, felt or ceramic blankets and tiles [107].

The never-completed X-33, and its envisioned full-scale successor,

7. Materials for Launch Vehicle Structures

VentureStar, were to have used an advanced metallic combination thermal protection system / aeroshell. It was to be constructed of titanium and Inconel. This represented a departure from the ceramic tile “acreage” TPS of the Space Shuttle and was meant to improve the durability of the vehicle. The metallic TPS was intended to be rain-proof, resistant to impact damage, and easily serviceable by replacing panels. However, the hottest areas of the structure, such as the nose and leading edges, were still planned to have carbon/carbon or carbon/silicon carbide panels [108].

In operation it was found that the Shuttle TPS was easily damaged and required careful observation and maintenance. This was known long before the *Columbia* failure, which can be seen as involving two separate TPS structural failures: one when foam insulation came loose from the External Tank, and another when the loose piece of insulation struck the carbon/carbon leading edge of the Orbiter’s wing, fatally damaging the ability of the wing to withstand the re-entry thermal environment. When the X-33 was developed, much effort was directed toward developing a more robust TPS.

Thermal protection systems are usually considered nonstructural, and are simply attached to the outer moldline of the structure. However, recent research has been done on load-carrying TPS, called integrated thermal protection systems.

Gogu *et al.* [109] compared materials for a corrugated core sandwich panel integrated TPS. They considered Ti-6Al-4V, zirconia, and an aluminosilicate/ceramic oxide fiber composite as web materials, aluminum, graphite-epoxy and vacuum hot-pressed beryllium as bottom facesheet materials, and Inconel 718, aluminosilicate/fiber and carbon/carbon as top facesheet materials. They concluded that using the aluminosilicate/fiber for the web and top facesheet, and beryllium as the bottom facesheet, led to a design only one-third the mass of the heaviest design.

Lindell *et al.* [110] described analysis and testing of an inflatable re-entry vehicle incorporating a flexible fabric-type thermal protection system consisting of layered polyimide and woven ceramic fabric. Because the structure was inflatable, it could be much larger than conventional re-entry vehicles (60-90 feet in diameter). A large surface area-to-weight ratio leads to lower heating and therefore less stringent requirements on the thermal protection system than would exist for other concepts

carrying the same mass.

Rakow and Waas [111] investigated an integral TPS consisting of actively cooled metal foam sandwich panels. The panels were composed of metal facesheets brazed to an open-cell metal foam core, with a coolant fluid circulated through the open-cell core structure. They discussed the advantages of this concept over previously considered actively cooled honeycomb core panels, which required separate coolant tubes to be built into the structure. The tubes do not permit as even a cooling effect as the metal foam. Henson [112] developed a class of continuum models for materials with small fluid-filled passages as may be used for active cooling.

Fesmire [113, 114] discussed the testing and potential applications of aerogel materials in LV thermal protection systems. Gels are materials that are mostly liquid by weight, but which have a crosslinked network that contributes enough rigidity that the material can support stress without flowing. An aerogel is a gel in which the liquid part has been replaced by a gas, resulting in a very low-density, porous material. Fesmire showed that aerogels are less prone to cryopumping than conventional foams, because of their high and finely dispersed porosity. Also, they are hydrophobic and therefore do not permit frost and ice to accumulate as do some other insulating materials.

Yao *et al.* [115] described the design and fabrication of a nickel-based superalloy honeycomb nonstructural thermal protection system for reusable applications. They measured the strength and thermal properties of the panel, and developed an oxidation-resistant coating containing a high-emittance layer for improved thermal performance.

7.9 Manufacturing Considerations

General references for this section: [116, 117, 118, 119, 120, 6, 121, 69, 122]

Introduction

Manufacturing of launch vehicles is a process that transforms raw materials into a space launch vehicle.⁸ This includes tanks, engines, structure and necessary sub-systems for full operations. This process has three phases;

⁸This section was contributed by Clyde S. Jones III, NASA Marshall Space Flight Center

7. Materials for Launch Vehicle Structures

fabrication, assembly, and checkout. Fabrication involves processing raw materials into the basic components for a launch vehicle. Examples of these components include commercially available metal plates and bars, fasteners, composite materials, adhesives, coatings, tubing, castings and forged metal. Assembly is the process by which components are collected from suppliers and assembled into complete systems. Most launch vehicle factories are primarily assembly facilities. Checkout is the process of verifying that the vehicle is ready for delivery. It is usually distributed during assembly, so that defects can be detected before too much value is added. A final checkout is performed as a last step before delivery to the launch site, and typically a functional or operational check.

Manufacturing Planning and Execution

Planning for manufacturing of space launch vehicles is similar in many ways to that in other industries. The size of components and types of materials are comparable to commercial aircraft, and quality standards share common approaches. Unique issues in launch vehicle manufacturing are primarily related to their low production rate and high cost. Even the most popular launch vehicles rarely exceed a production rate of one unit per month, and most are produced at less than half that rate. In contrast, the Boeing 747, for instance, with a similar size and complexity, is produced at a rate of one to six per month. Compared to other commercial manufacturing, the comparison is even more pronounced. The automobile industry may produce one thousand vehicles in a shift, and each vehicle is far less valuable.

The significance of this difference in production rate is manifested in several ways. If the production process for a particular component or assembly is only performed a few times in a year, there will be a stronger reliance on written procedures to assure that the part is produced correctly. A space launch vehicle has a greater cost per component, so that each processing step is financially riskier than in mass production.

With large, expensive components, and precise fit-up tolerances, tooling to position the components can be very complex. Manufacturing simulation computer systems are used to help optimize the flow of large assemblies through the factory. As the cost of computing power declines, simulation systems are an economical way to analyze different manufacturing scenarios and iterate an optimum flow. These systems can then

use design information to program robots, machine tools, and welding systems for very complex assemblies.

Simulation systems can adjust the programs of large complex machines to fit unique model configurations and even compensate for some types of geometric imperfections in components. Fabrication of an aluminum-phenolic sandwich structure for the common bulkhead on the Ares Upper Stage demonstrated how manufacturing simulation systems could match two welded aluminum domes with their phenolic sandwich material. While the welded domes had small areas that did not match the design within the tolerance required to complete the adhesive joints, computer systems match-machined the phenolic to fit the imperfect parts and successfully completed the adhesive bonded assembly.

A successful manufacturing planning system will provide for tracking the use of different materials and components to allow traceability in the case of defects. If the certification of any particular lot of parts or material used in manufacturing comes into question at any time, the manufacturing planning system can determine where the questionable parts were used on any vehicle, allowing replacement, or even acceptance by further testing or analysis. In such situations, accurate information on the pedigree of any part or material used on the vehicle can be invaluable.

Nonconformances, meaning processes that were carried out differently than the design intent, are bound to occur, so a process for disposition is necessary. Some nonconformances are acceptable. A Material Review Board develops and documents the disposition of a nonconforming part or process. A typical process is: discovery of the nonconformance, documentation of the technical details and application, determination as to whether corrective action is needed, and if necessary development of a corrective action.

Assembly Processes: Welding

Welding is the primary assembly method for large cryogenic tanks. A pressurized tank using welded joints can reduce dry mass and part count compared to a mechanical joint, and is less likely to leak over a wider range of operating conditions. Disadvantages include the requirement of a high skill level for production workers, and the cost of non-destructive inspection processes to screen for cracks or related defects. Historically, welding has been a critical technical and schedule driver in production of

7. Materials for Launch Vehicle Structures

launch vehicles [39].

Welding aluminum for launch vehicle tanks and structures has been a well-proven process since the 1960s. Aluminum alloys commonly used include 2219, 2014, 2024, and 2195. These alloys have the advantage of high specific strength and good fracture toughness at cryogenic temperatures. An important feature of aerospace aluminum alloys used for manufacturing launch vehicles is that they have better fracture properties at cryogenic temperatures, so that a less expensive room temperature acceptance test is sufficient. Aerospace aluminum alloys exhibit lower mechanical properties in the weld joint than areas unheated by welding due to oxide trapped as the metal solidifies, and cracking as the metal cools. Strength reduction can be mitigated by adding extra thickness at the weld joint.

The weld process is usually developed to concentrate the heat as much as practical, allowing higher welding speeds. This reduces the heat-affected zone, minimizing heat effects on the base material temper.

Welds on a launch vehicle structural element are usually made automatically rather than manually. This results in more consistent heat input along the weld joint. This consistency makes the weld properties more predictable, and reduces distortion. Over the years, advances in computing hardware and software have made automatic welding systems more consistent over a wider variety of production conditions. In the 1960s, and during the first few builds of the Space Shuttle External Tank, electronic servocontrols with operational logic provided by relays were the norm for welding automation. By the mid-1980s, digital computers were commonplace for automation, improving the operator interface, and providing more accurate adjustment of all weld parameters that affect the quality of the process. A very important improvement by digital control systems was detailed recording of parameters as the weld progressed. Computers have allowed for precise programming of each parameter before welding starts, allowing the welding engineer to build a successful scheme for each joint. As welding progresses, the computer records each parameter multiple times each second. The result is detailed data on each weld, which can be compared with previous attempts to fine tune the procedure.

Robotic welding was introduced for launch vehicle applications in the late 1980s. Robots apply the consistency of welding automation to joints

with complex curvature. The programmable path of the robot can reduce the cost of motion control compared to a specially designed system for a specific geometry. Robots using the gas tungsten arc welding (GTAW) process were able to join a wide variety of components previously welded manually on the Space Shuttle Main Engine in 1989, and are still in use today. A robot using the plasma arc process was used for a saddle joint on the docking nodes of the International Space Station in the mid-1990s. The robot used eight axes of motion to position both the component and the weld torch in the ideal orientation for a successful weld. Because the robot could be programmed for multiple paths, it was also used for other welds on the Space Station structure, avoiding the need for additional welding systems. Currently the Orion crew vehicle uses one robot to perform friction stir welding for every weld joint on the vehicle, including circumferential and linear geometries. Using a robot to bring the welding process to multiple fixtures and weld stations reduces the overall floor space that would have been required for conventional welding. The universal programmability feature inherent in the robot is ideal for low production rate of launch vehicles, providing a cost-effective approach to design changes and different model configurations.

Many different welding processes have been used successfully in a production system on operational launch vehicles, including gas metal arc, resistance, GTAW, plasma arc, electron beam (EB), and friction stir welding processes. Gas metal arc has been phasing out since the 1960s because it is prone to porosity and oxide inclusions when welding aluminum. When used on the Saturn vehicles, the process required significant rework compared to the welding processes used today [39].

Resistance welding processes have been used extensively on launch vehicles. It worked well with the 301 and 321 stainless steels used in the Atlas family, and is still used for the Centaur upper stage. This process has not found similar success in aluminum structures, primarily due to inconsistent quality. This is likely due to the high resistance of aluminum oxide that quickly forms on the surface of aluminum, affecting the current flowing between the electrodes. The overlap design of a resistance-welded joint also leads to difficulties in applying non-destructive inspection techniques. Other applications for this welding process include structural covers for insulation systems.

GTAW is still commonly used on aluminum welds for launch vehicles.

7. Materials for Launch Vehicle Structures

These processes have higher energy density than the gas metal arc process, resulting in a smaller heat affected zone and thus higher mechanical properties. GTA and plasma welding processes can be operated in alternating polarity, which provides a cathodic cleaning action to aluminum during welding. This reduces the presence of oxides, minimizing the chance for oxide inclusions in the weld zone, and improves flow of the molten pool. Oxides are further discouraged in the weld zone by abrading the joining surfaces, through draw filing, wire brushing, or other mechanical means. Since aluminum will develop a surface oxide quickly, there is usually a time limit established between completion of surface cleaning and when welding starts. If this limit is exceeded, additional cleaning is required before welding.

Electron beam welding uses a high voltage to accelerate electrons, which are focused using magnetic fields to melt metals for welding. This welding process has the advantage of very high energy density, which can penetrate and join thick parts with minimal distortion, and minimal effect on the temper of adjacent material. It is used extensively on launch vehicles to assemble engine components, hermetically seal batteries and join thick materials used in heavily loaded structural parts. The process takes place in a vacuum, so metals that oxidize at elevated temperatures, such as titanium, can be welded with minimal risk of included oxides. Since the process must take place in a vacuum chamber, there is a practical limit to the size of components that can be EB welded. It is also limited to metals that are non-magnetic, that won't deflect the beam during welding.

Friction stir welding has been adopted by launch vehicle manufacturers rapidly since its invention in the early 1990s. FSW is ideally suited for aluminum, because it is relatively soft at elevated temperatures. This allows commonly available tool steels to be used for the pin that applies friction to the part. It also reduces the forces that must be reacted by the weld tooling. While titanium and ferrous alloys have been welded with the FSW process, aluminum alloys are the most common application. The first application of this process in a production environment was in Europe, fabricating aluminum structures for shipbuilding in the mid-1990s, applied to a 6000-series alloy.

The first launch vehicle application was by Boeing on a Delta II variant that first flew in 2001, which applied the process to the 2024 alloy

on longitudinal welds. Lockheed Martin and NASA developed a more complex application for longitudinal barrel welds on the Space Shuttle External Tank in the early 2000s. The External Tank used Al-Li 2195 for these parts, and the weld joints tapered in thickness from almost 16 mm at the LH₂ tank aft dome, down to 8 mm at the LH₂ tank forward dome. This application required a more sophisticated pin tool that could adjust its extension as the weld traveled along the joint. An automated method to control the pin extension was developed to maintain the proper depth and stir the weld completely through the weld joint thickness. The last five External Tanks produced took advantage of this new technology. The Delta IV launch vehicle was designed with FSW in mind. All the longitudinal welds were joined using FSW, while circumferential welds used a version of variable polarity plasma arc. The design of the LOX and LH₂ tanks eliminated some circumferential welds by increasing the number of barrel panels and longitudinal welds

Weld Distortion

A common problem in all welding processes is distortion. A distorted component is more difficult to join to adjacent structure, and has higher residual stress, both of which reduce structural efficiency.

Distortion resulting from the weld process comes primarily from shrinkage in the weld zone, but can also result from the interaction of residual stresses in each component, and how they change after welding heats the parts. Because high-strength materials are often used in launch vehicles, distortion is exacerbated since localized shrinkage in the weld area is not distributed across a larger area by yielding. There are a variety of mitigation techniques for weld distortion and fit-up issues. Well-designed fixtures position the parts precisely, and pneumatic actuators restrain the parts during application of heat. (Hydraulic actuation is rarely used for welding fixtures to avoid contamination by leaking fluid.) Alignment is measured before welding, and extra pressure is applied, or trimming operations are used, to bring the fit-up within specifications. Tack welds can be used to restrain the parts and maintain alignment as heat is applied. Spacing, depth of penetration and the sequence of application are all important parameters in tack welding.

Weld processes with low energy density and a less concentrated heat source, such as gas metal arc, are usually more prone to distortion. Areas

7. Materials for Launch Vehicle Structures

with thinner material, or areas that require more heat passes, exhibit more distortion. Weld repair areas are more prone to excessive distortion, because the part is subjected to multiple weld passes and solidification shrinkage in repair areas. The additional heat reduces the strength of the base metal by changing any previous tempering processes. Multiple welds in the same area will also act on any residual stresses in the components being joined, producing additional distortion and residual stress.

High-energy-density weld processes such as EB and laser welding result in less distortion. Resistance welding, plasma arc, and GTAW fit between these two extremes. This is primarily due to a smaller molten pool along the weld seam, which reduces metal solidification shrinkage. FSW produces less shrinkage because it does not melt the material.

After welding is completed, procedures typically require measurement of the joint geometry to verify that reinforcement, peaking, and offset (or mismatch) are within specification. If corrective measures are warranted, planishing can be used to compress the weld reinforcement and correct geometry problems. In rare cases, additional welding passes can be used to shrink certain areas to bring the geometry into compliance. This approach is less often used because of the risk of distortion.

Mechanical Assembly Processes

Mechanical fastening systems are well-developed for use on launch vehicles. Major structural elements are joined using bolts and related fasteners with precision, accuracy and predictable mechanical properties. While the pressurized components of launch vehicle tanks are more typically welded, mechanically fastened components are used for propellant feedline attachments, venting components, personnel access covers and instrumentation feed-throughs. Bolted connections allow disassembly and reassembly. Keys to success with bolted joints include good fit and adequate fastener torquing. Success is verified by measuring torque on the fastener and a leak test. If fasteners are to be threaded to blind holes in an aluminum structure, a threaded insert is normally used. In aerospace applications, threaded fasteners require at least one locking device to prevent loss of preload, and lock wire is typical for this application. Thread locking compounds are not commonly used due to temperature extremes experienced on launch vehicles, but thread sealing compound has been used on the Space Shuttle External Tank to reduce infiltration of liquid nitrogen

behind thermal protection foam.

Riveting has been used in unpressurized structures of launch vehicles such as the Intertank subassembly of the Space Shuttle External Tank.

7.10 Summary, Trends and Outlook

Preparing for the launch of an expensive, specialized payload on an expendable vehicle involves “good practice” processes that do not always have a firm scientific basis. Low flight rates make it difficult to rationally assess the costs and benefits of analysis, testing, and quality control. The verification criteria, qualification strategies, and analysis methods that have matured over the past few decades have been described here.

Space launch vehicles utilize many of the same materials as aircraft: the 2000-, 6000- and 7000-series aluminum alloys, laminated and filament-wound composites, high-strength steels, and titanium alloys. The need for mass efficiency is the primary driver for both aircraft and LVs. However, the frequent use of cryogenic propellants, as well as high aerodynamic heating environments, impose challenging thermal conditions on LVs. On the other hand, the short lifetime of expendable launchers reduces the importance of fatigue, fracture, corrosion resistance and other properties governing long-term material behavior. For reusable vehicles, fatigue and fracture can be just as important as in aircraft, and the design of a robust, reusable thermal protection system for atmospheric reentry requires all materials and structures technology to be brought to bear.

Most of the material processing and joining technologies used in aircraft are also used in launch vehicles. Welding is a key technology in LV structures. Friction stir welding is arguably the most significant advance in the state of the art of materials and structures since the development of composites.

Aluminum-lithium alloys, now introduced on a large scale in the Space Shuttle External Tank, represent a significant improvement in strength-to-weight ratio over conventional aluminum alloys. Composite propellant tanks can offer further gains in mass efficiency with judicious design, but the need for robust joints and minimization of permeation after fatigue remain significant roadblocks to the use of composites in pressurized structure. However, filament-wound composite solid rocket motor cases are a mature and widespread technology.

7. Materials for Launch Vehicle Structures

Composites continue to be an active area of research. Bolted and bonded composite-to-composite or composite-to-metal joints present challenges to both design and analysis. Textile preforms, new methods of curing and new matrix materials are all pathways to meeting these challenges. New materials such as aerogels, metal foam and nanocomposites can be fabricated and tested at the laboratory scale; these materials may soon find applications in production. Another technology enabling the wider use of composites is rigorous methods for predicting gradual progression of damage and assessing residual strength.

Looking further into the future, nanostructured materials such as carbon nanotubes and graphene sheets appear to hold great promise. These materials have interesting electrical and thermal properties as well as extremely high specific strength and stiffness. Current research seeks to reduce the cost of producing such materials and to assemble them in quantities usable for structural applications. Modifying current materials such as polymeric matrix materials for composites by the addition of nanostructured materials may be a significant first step in their more widespread use (see Chapter 3 for details.) A system study predicted a factor of two improvement in weight if conventional carbon fiber composites were used throughout a structure, but a factor of ten improvement if projected properties of carbon fiber nanotube reinforced materials could be realized [123].

Advanced materials identified in [123] and potentially applicable to launch vehicle structures included:

- Titanium-aluminum alloy
- Alumina fiber/aluminum matrix composite
- Aluminum and titanium alloy foam as core materials for sandwich structures
- Aluminum-beryllium alloys
- Silicon carbide fiber/beryllium matrix composite
- Carbon nanotube fiber/aluminum matrix composite
- Single-crystal metals, nanotube-reinforced alloys and new superalloys for high-temperature applications
- Ceramic matrix composites

Bionics or biomimetics [124] is another material and structural concept that is a current topic of research. It has long been realized that if a

structure were capable of large-scale adaptations, it could be optimized for two or more very different environments and therefore be much more efficient than a one-size-fits-all design. Flaps, slats and trim tabs may be regarded as first steps down this path. Swing-wings and deployable space structures display yet more adaptation, but these continue to use conventional materials. A bionic structure would incorporate flexible skin materials capable of large strains, as well as internal bracing akin to a skeleton and actuators akin to muscles. Integral fluid passages could provide both thermal control and the ability to change the shape or stiffness of the structure by changing pressures or flow rates. Such concepts could answer requirements for extremely efficient, adaptable, robust launch vehicle structures in future reusable, SSTO systems

Bibliography

- [1] Anonymous, "Aerobee Fired in Reusability Test," *Goddard News*, August 1976.
- [2] Filin, V. M., "Energia Launch Vehicle: Development History and Main Achievement," *Acta Astronautica*, Vol. 43, No. 1-2, 1998.
- [3] Anon, *Annual Survey of Manufactures*, Government Printing Office, Washington, D.C., 2006.
- [4] Hammond, W. E., *Space Transportation: A Systems Approach to Analysis and Design*, AIAA Education Series, AIAA Press, Reston, Virginia, 1999.
- [5] Kyle, E., "<http://www.spacelaunchreport.com>," Website, Retrieved in June 2009.
- [6] Chang, I.-S., "Investigation of Space Launch Vehicle Catastrophic Failures," *AIAA J. Spacecraft and Rockets*, Vol. 33, No. 2, 1996, A catalog and analysis of worldwide space launch vehicle failures for the period 1984 to 1994.
- [7] Wilhite, A. W., Englund, W. C., Stanley, D. O., Naftel, J. C., Lepsch, R. A., Bush, L. B., and Wurster, K. E., "Technology and Staging Effects on Two-Stage-to-Orbit Systems," *J. Spacecraft and Rockets*, Vol. 31, No. 1, January 1994, pp. 31, Evaluation of structural designs for horizontal takeoff and landing two-stage systems.

Bibliography

- [8] FAA, "Metallic Materials Properties Development and Standardization," Handbook MMPDS-02, FAA, Washington, D.C., April 2005.
- [9] Anon, "Composite Materials Handbook, Revision G," online handbook, 2009.
- [10] Hart-Smith, L. J., "The Role of Biaxial Stresses in Discriminating Between Meaningful and Illusory Composite Failure Theories," *Proceedings of Ninth DOD-NASA-FAA Conference on Fibrous Composites in Structural Design*, Vol. 3, FAA, 1992, pp. 1507–1528.
- [11] Dawicke, D. S. and Pollock, W. D., "Biaxial Testing of 2219-T87 Aluminum Alloy Using Cruciform Specimens," Contractor Report 4782, NASA, Hampton, Virginia, 1997.
- [12] Chen, P. S. and Stanton, W. P., "A New Aging Treatment for Improving Cryogenic Toughness of the Main Structural Alloy of the Super Lightweight Tank," Technical Report TM-1996-108524, NASA Marshall Spaceflight Center, Huntsville, Ala., 1996, Technique for improving fracture properties of Al-Li alloy used on Space Shuttle SLWT.
- [13] Muraca, R. F. and Whittick, J. S., "Materials Data Handbook: Stainless Steel Type 301," Tech. rep., NASA, Marshall Space Flight Center, Ala., April 1972.
- [14] Key, C. F., "Materials Selection List for Space Hardware Systems," Tech. Rep. MSFC-HDBK-527, Marshall Space Flight Center, Marshall Space Flight Center, Ala., September 1988.
- [15] Anon, *Aerospace Structural Metals Handbook*, CINDAS LLC, West Lafayette, Indiana, Updated annually.
- [16] Kraemer, R. S. and Wheelock, V., *Rocketdyne: Powering Humans into Space*, AIAA, 2005.
- [17] Gutman, I. L. and Kulikova, G. V., "BURAN Orbital Spaceship Airframe Creation: Non-Metallic Structural Materials of the BURAN Orbiter," <http://www.buran-energia.com/documentation/documentation-akc-non-metallic-material.php>, Retrieved in December 2009.

- [18] Anon, "Fracture Control Requirements for Payloads using the Space Shuttle," NASA Standard 5003, October 1996.
- [19] Gerard, G., "Optimum Structural Design Concepts for Aerospace Vehicles," *J. Spacecraft*, Vol. 3, No. 1, January 1966, pp. 5, A comprehensive bibliography and review of the 1966 state of the art in analytical optimum design of aerospace vehicle structures.
- [20] Mukhopadhyay, V., "A Conceptual Aerospace Vehicle Structural System Modeling, Analysis and Design Process," *Proceedings of the 48th AIAA Structures, Structural Dynamics and Materials Conference*, AIAA, AIAA, Honolulu, 2007.
- [21] Blair, J. C. and Ryan, R. S., "Role of Criteria in Design and Management of Space Systems," *AIAA J. Spacecraft and Rockets*, Vol. 31, No. 2, 1994.
- [22] Guikema, S. D. and Paté-Cornell, M. E., "Bayesian Analysis of Launch Vehicle Success Rates," *J. Spacecraft and Rockets*, Vol. 41, No. 1, January-February 2004, pp. 93, Bayesian approach to predicting launch vehicle reliability with emphasis on vehicles with little flight history.
- [23] Miyazawa, M., "Homegrown technology for Japan's heavy-lift launcher," *IEEE Spectrum*, Vol. 26, No. 3, March 1989, pp. 51-55.
- [24] Anon, "Eastern and Western Range Safety Policies and Procedures," Tech. Rep. EWR 127-1 Rev. B, Range Safety Office, 45th Space Wing and 30th Space Wing, U.S. Air Force, Patrick AFB, Florida, April 2001.
- [25] Anon, *Range Safety User Requirements Manual*, Air Force Space Command, Peterson AFB, Colo., July 2004.
- [26] Anon, *Launch Safety Requirements for Air Force Space Command Organizations*, Air Force Space Command, Peterson AFB, Colo., February 2007.
- [27] Hannum, J. A. E., "Hazards of Chemical Rockets and Propellants," Tech. Rep. 394, Chemical Propulsion Information Agency, Baltimore, June 1985.

Bibliography

- [28] Anon, *Standard General Requirements for Safe Design and Operation of Pressurized Missile and Space Systems*, United States Department of Defense, MIL-STD-1522, Los Angeles, 1984.
- [29] Anon, "Streamlining Space Launch Range Safety," Tech. rep., National Research Council, Committee of Space Launch Range Safety, Washington, D.C., 2000.
- [30] Sarafin, T. P., editor, *Spacecraft Structures and Mechanisms: From Concept to Launch*, Space Technology Library, Microcosm Press/Springer, Hawthorne, Calif, 1995.
- [31] Whitehair, C. L. and Wolfe, M. G., "The New U.S. Government Space System Acquisition Policy and its Impact of Risk Management Procedures," *Acta Astronautica*, Vol. 40, No. 2-8, 1997.
- [32] Gregory, F. D., "Safety and Mission Assurance in a Better, Faster, Cheaper Environment," *Acta Astronautica*, Vol. 39, 1996.
- [33] Swanson, D., Matsunaga, S., and Lollock, R., "Mission Assurance: "Baking In" Mission Success," *High Frontier*, Vol. 6, No. 1, 2009.
- [34] Weingarten, V. I., Seide, P., and Peterson, J. P., *Buckling of Thin-Walled Circular Cylinders*, NASA, Marshall Space Flight Center, Alabama, 1965, revised 1968, Analysis formulas, including knock-down factors, for buckling of thin-walled shells.
- [35] Bushnell, D., "Buckling of Shells - Pitfall for Designers," *AIAA Journal*, Vol. 19, No. 9, September 1981, pp. 1183, Discursive review of the shell buckling problem and its relevance to launch vehicle design.
- [36] Hilburger, M. W., "Developing the Next Generation Shell Buckling Design Factors and Technologies," *Proceedings of the 53rd AIAA/ASME/ASCE/AHS/ASC Structures, Structural Dynamics and Materials Conference*, No. AIAA-2012-1686, AIAA, AIAA, Honolulu, April 2012.
- [37] Anon, *Test Requirements for Launch, Upper-Stage and Space Vehicles*, United States Department of Defense, MIL-HDBK-340A, Los Angeles, rev. a ed., 1999.

- [38] Flynn, T. M., *Cryogenic Engineering*, CRC, 2nd ed., 2004, Chapter 11 covers material selection for cryogenic tanks.
- [39] Bilstein, R. E., *Stages to Saturn: A Technological History of the Apollo/Saturn Launch Vehicles*, No. SP-4206 in NASA History, NASA, Washington, D. C., 1996.
- [40] Bruhn, E. F., *Analysis of Missile Structures*, Tri-State Offset Company, 1967.
- [41] Bert, C. W., Mills, E. J., and Hyler, W. S., "Mechanical Properties of Aerospace Structural Alloys Under Biaxial-Stress Conditions," Technical Report TR-66-229, Battelle Memorial Institute - Columbus Laboratories, Air Force Materials Laboratory, Wright-Patterson AFB, Ohio, 1966.
- [42] Shradler, J., *MSFC-STD-1249, NDE Guidelines and Requirements for Fracture Control Programs*, NASA, Marshall Space Flight Center, Alabama, 1985.
- [43] Mynors, D. J. and Zhang, B., "Applications and capabilities of explosive forming," *J. Materials Processing Technology*, Vol. 125-126, September 2002, pp. 1–25, Explosive forming experience including space launch applications.
- [44] Slysh, P., "Isogrid Structures Technology and Applications," Technical Report SAWE CAS-2, General Dynamics Convair Division, Los Angeles, 1978.
- [45] Birt, M. J., Domack, M. S., Hafley, R. A., and Pollock, W. D., "Characterization of Al-Cu-Li Alloy 2090 Near Net Shape Extrusion," Technical Report TM-1998-207668, NASA Langley Research Center, Hampton, Va., 1998, Properties of skins and stiffeners fabricated from Al-Cu-Li extrusions for space applications.
- [46] Meyer, R. R., Harwood, O. P., and Orlando, J. I., "Isogrid Design Manual," Tech. Rep. CR-124075, NASA, Marshall Space Flight Center, Alabama, February 1973.
- [47] Mendez, P. F. and Eagar, T. W., "Welding Processes for Aeronautics," *Advanced Materials & Processes*, May 2001, pp. 39–43, Covers space launch applications as well.

Bibliography

- [48] Anon, "Effects of Low Temperatures on Structural Metals," Tech. Rep. SP-5012, NASA, December 1964.
- [49] Hunsicker, H. Y. and Hess, J. H., "New Weldable High Strength Aluminum Alloys for Cryogenic Service," Technical Report N66-21814, NASA Marshall Spaceflight Center, Huntsville, Ala., 1966, Overview of aluminum alloys for cryogenic tanks, including weldability and strength properties.
- [50] Sessler, J. and Weiss, V., "Materials Data Handbook: Aluminum Alloy 2219," Tech. rep., NASA, Marshall Space Flight Center, Ala., March 1966.
- [51] Pessin, M. A., "Lessons Learned from Space Shuttle External Tank Development: A Technical History of the External Tank," Technical report, NASA, Marshall Space Flight Center, Alabama, 2002, Discussion of materials, processes and structures problems and solutions encountered during development of all variants of the Shuttle External Tank.
- [52] Muraca, R. F. and Whittick, J. S., "Materials Data Handbook: Aluminum Alloy 2014," Tech. rep., NASA, Marshall Space Flight Center, Ala., March 1972.
- [53] Davis, R. A., "Evaluation of Welded 2219-T87 Aluminum Alloy," Technical report, NASA, Marshall Spaceflight Center, Ala., 1963, Strength of MIG and TIG welds on 2219 alloy for Saturn SI-C booster.
- [54] Wagner, W. A., "Liquid Rocket Metal Tanks and Tank Components," Space Vehicle Design Criteria SP-8088, NASA Glenn Research Center, Cleveland, 1974.
- [55] Mendez, P. F. and Eagar, T. W., "New Trends in Welding in the Aeronautic Industry," *Proceedings of the 2nd Conference of New Manufacturing Trends*, Bilbao, Spain, 2002.
- [56] Cao, G. and Kou, S., "Friction Stir Welding of 2219 Aluminum: Behavior of θ (Al_2Cu) particles," *Welding Research supplement to the Welding Journal*, January 2005, Micrographic study showing that friction stir welding does not cause liquation of Alloy 2219.

- [57] Hales, S. J. and Wagner, J. A., “Superplastic Forming of Al-Li Alloys for lightweight, low-cost structures,” Technical Report N91-24065, NASA Langley Research Center, Hampton, Va., 1991.
- [58] Kirn, E. P., Graf, N. A., and Ely, K. W., “LOX-Compatible Composite Tank for Aerospace Applications,” United States Patent 6837464, 2005.
- [59] Anon, *Reusable Launch Vehicle: Technology Development and Test Program*, National Academy Press, 1995, Policy-level overview of material selection and structural design procedures for the ill-fated X-33 reusable single-stage-to-orbit vehicle.
- [60] Bhat, B. N., “Materials, Processes and Manufacturing in Ares I – Upper Stage: Integration with Systems Design and Development,” *Proceedings of the National Space and Missile Materials Symposium*, NASA, Las Vegas, Nevada, June 2008.
- [61] Pope, D. H. and Penner, J. E., “Testing of Polymeric Materials for Use in Multiple-Ply Expulsion Bladders for Cryogenic Liquids,” *AIAA J. Spacecraft*, Vol. 5, 1968.
- [62] Gleich, D. and L’Hommedieu, F., “Metallic Bladders for Cryogenic Fluid Storage and Expulsion Systems,” *AIAA J. Spacecraft*, Vol. 5, No. 9, 1968.
- [63] Calabro, M., Bonnal, C., and Lenstch, A., “ELV: Pressure-fed LOX/LH2 upper stage,” *Acta Astronautica*, Vol. 64, March 2009, pp. 1015–1020.
- [64] Cataldo, C. E., “Compatibility of Metals with Hydrogen,” Technical Memorandum X-53807, NASA, Marshall Space Flight Center, Ala., 1968.
- [65] Martin, R. E., “The Atlas and Centaur “Steel Balloon” Tanks: A Legacy of Karel Bossart,” *Proceedings of the 40th Congress of the International Astronautical Federation*, No. IAA-89-738, International Astronautical Federation, International Astronautical Federation, Malaga, Spain, October 1989, Development history of the steel balloon tanks, written by an early participant.

Bibliography

- [66] Sankar, B., Xu, J., and Pelt, J. V., "Lightweight Composite Tanks for Liquid Hydrogen Storage," Tech. rep., NASA, Cleveland, Ohio, 2008, Hydrogen permeability study of composites via microcracks.
- [67] Morino, Y., Shimoda, T., Morimoto, T., Ishikawa, T., and Aoki, T., "Applicability of CFRP materials to the cryogenic propellant tank for reusable launch vehicle (RLV)," *Adv. Composite Mater.*, Vol. 10, No. 4, 2001, pp. 339–347.
- [68] Cast, J., "Composite Hydrogen Tank Test Completed for DC-XA," NASA Headquarters press release 96-13, January 1996.
- [69] Harris, C. E., Starnes, J. H., and Shuart, M. J., "An Assessment of the State-of-the-Art in the Design and Manufacturing of Large Composite Structures for Aerospace Vehicles," Report TM-2001-210844, NASA, Hampton, Virginia, 2001, A comprehensive survey of composite primary structures for commercial fixed-wing aircraft, rotorcraft, and space launch vehicles.
- [70] Hartunian, B., "McDonnell Douglas Composite Cryotanks - A Personal History," *High-Performance Composites*, 2006.
- [71] Robinson, M., "Composite Structures on the DC-XA Reusable Launch Vehicle," *Journal of Advanced Materials*, Vol. 28, No. 3, 1997, pp. 9–18.
- [72] Heydenreich, R., "Cryotanks in future vehicles," *Cryogenics*, Vol. 38, No. 1, January 1998, pp. 125–130, Discusses carbon-fiber reinforced plastic (composite) tanks for liquid oxygen and hydrogen storage in launch vehicles.
- [73] Graf, N. A., Schieleit, G. F., and Biggs, R., "Adhesive Bonding Characterization of Composite Joints for Cryogenic Usage," NASA, NASA, Boston, 2000.
- [74] Miller, S. G. and Meador, M. A., "Polymer-Layered Silicate Nanocomposites for Cryotank Applications," *Proceedings of the 48th AIAA/ASME/ASCE/AHS/ASC Structures, Structural Dynamics and Materials Conference*, AIAA, AIAA, Honolulu, April 2007, Characteristics of layered silicate nanocomposites in cryotank applications.

- [75] Pavlick, M. M., Johnson, W. S., Jensen, B., and Weiser, E., "Evaluation of mechanical properties of advanced polymers for composite cryotank applications," *Composites: Part A*, Vol. 40, 2009, pp. 359–367.
- [76] Black, S., "An update on composite tanks for cryogenics," *High-Performance Composites*, 2005.
- [77] Whitfield, H. K., "Solid Rocket Motor Metal Cases," Space Vehicle Design Criteria SP-8025, NASA Glenn Research Center, Cleveland, 1970.
- [78] Peters, S. T., Humphrey, W. D., and Foral, R., *Filament Winding Composite Structure Fabrication*, Society for the Advancement of Material and Process Engineering, Covina, Calif., 2nd ed., 1999, Overview of the filament winding process for building composite rocket motor cases, pressure vessels, fuselage sections and other aerospace structures.
- [79] Poe, C. C., "Summary of a Study to Determine Low-Velocity Impact Damage and Residual Tension Strength for a Thick Graphite-Epoxy Motor Case," Technical Memorandum TR-102678, NASA Langley Research Center, Hampton, Virginia, 1990.
- [80] Gargiulo, C., Marchetti, M., and Rizzo, A., "Prediction of Failure Envelopes of Composite Tubes Subjected to Biaxial Loadings," *Acta Astronautica*, Vol. 39, No. 5, 1996.
- [81] Bert, C. W. and Hyler, W. S., "Design Considerations in Material Selection for Rocket Motor Cases," *AIAA J. Spacecraft and Rockets*, Vol. 4, No. 6, 1967.
- [82] Alexander, R. V. and Fournier, C. A., "Cost and Performance Considerations in the Selection of Structural Materials for Ultra-Large-Size Booster Motors," *AIAA J. Spacecraft*, Vol. 1, No. 1, 1963.
- [83] Pionke, L. J. and Garland, K. C., "Evaluation of Shuttle Solid Rocket Booster Case Materials," Contractor report, NASA, Marshall Space Flight Center, Ala., 1973.

Bibliography

- [84] Daniels, C. M., "Liquid Rocket Lines, Bellows, Flexible Hoses, and Filters," Space Vehicle Design Criteria SP-8123, NASA Glenn Research Center, Cleveland, 1977.
- [85] Nettles, A. T., "Impact Damage Resistance of Carbon/Epoxy Composite Tubes for the DC-XA Liquid Hydrogen Feedline," Technical Paper 3583, NASA, Marshall Space Flight Center, Ala., 1995.
- [86] Thesken, J. C., Murthy, P., and Phoenix, L., "Composite Overwrap Pressure Vessels: Mechanics and Stress Rupture Lifting Philosophy," *Proceedings of the 48th AIAA/ASME/ASCE/AHS/ASC Structures, Dynamics and Materials Conference*, AIAA, Honolulu, 2007.
- [87] Saulsberry, R., Greene, N., Cameron, K., Madaras, E., Grimes-Ledesma, L., Thesken, J., Phoenix, L., Murthy, P., and Revilock, D., "Nondestructive Methods and Special Test Instrumentation Supporting NASA Composite Overwrapped Pressure Vessel Assessments," *Proceedings of the 48th AIAA/ASME/ASCE/AHS/ASC Structures, Dynamics and Materials Conference*, AIAA, Honolulu, 2007.
- [88] Anon, "Composite Overwrapped Pressure Vessels (COPVs)," AIAA Standard for Space Systems, 2006.
- [89] Armstrong, H. H., Burns, A. B., and Crawford, R. G., "Large Stiffened Cylinder Weight-Cost Comparisons between Be-38% Al and Other Materials," *AIAA J. Spacecraft*, Vol. 4, No. 3, 1967.
- [90] Vosteen, L. F. and Hadcock, R. N., "Composite chronicles; A study of the lessons learned in the development, production and service of composite structures," Report CR-4620, NASA, 1994.
- [91] Kobayashi, M., Sakai, S., and Shimizu, R., "Development of H-2A launch vehicle composite interstage structure," *Advanced Composite Materials*, Vol. 10, No. 2,3, 2001, pp. 247–253.
- [92] Stalder, J. R., "The Useful Heat Capacity of Several Materials for Ballistic Nose-Cone Construction," Technical note TN-4141, Ames Aeronautical Laboratory, National Advisory Committee for Aeronautics, Washington, 1957.
- [93] Radcliffe, R. L., "Surveyor Nose Fairing Mode Survey," Contractor Report CR-54524, General Dynamics / Astronautics Division, San

- Diego, 1964, Contains design description of an early launch vehicle payload fairing.
- [94] Butcher, J. R., "The Separation Test in Vacuum of the Ariane 4 Payload Fairing," *Proceedings of the 15th Space Simulation Conference*, NASA Goddard Spaceflight Center, 1988.
- [95] Duret, F., "Small and Medium Launch Vehicles to Extend the Ariane 5 Transportation Service," *Acta Astronautica*, Vol. 40, No. 2-8, 1997, Design descriptions of several Russian launchers.
- [96] Schwingel, D., Seelinger, H.-W., Vecchionacci, C., Alwes, D., and Dittrich, J., "Aluminium foam sandwich structures for space applications," *Acta Astronautica*, Vol. 61, 2007, pp. 326–220, An aluminum foam-core sandwich structure is manufactured and tested.
- [97] Lane, S. K., Henderson, K., Williams, A., and Ardelean, E., "Chamber Core Structures for Fairing Acoustic Mitigation," *AIAA J. Spacecraft and Rockets*, Vol. 44, No. 1, 2007.
- [98] Ochinerio, T., Deiters, T., Higgins, J., Arritt, B., Blades, E., and III, J. N., "Design and Testing of a Large Composite Asymmetric Payload Fairing," *Proceedings of the 50th AIAA/ASME/ASCE/AHS/ASC Structures, Structural Dynamics and Materials Conference*, AIAA, AIAA, Palm Springs, Calif., May 2009.
- [99] Scolamiero, L., "Modeling of Honeycomb Core for Sandwich CTE Predictions," *Proceedings of the International Conference on Spacecraft Structures and Mechanical Testing*, Noordwijk, Netherlands, 1991.
- [100] Kaplan, A., "Debonding of Graphite Sandwich Panels," *Proceedings of the 43rd International SAMPE Symposium*, 1998.
- [101] Anon, *Insert Design Handbook*, No. PSS-03-1202, European Space Agency, 1987.
- [102] Anon, *Handbook for the Montreal Protocol on Substances That Deplete the Ozone Layer*, Ozone Secretariat, United Nations Environment Programme, 7th ed., International treaty agreements limiting the use of ozone-depleting substances such as foam insulation blowing agents.

Bibliography

- [103] Noever, D. A., Cronise, R. J., Wessling, F. C., McMannus, S. P., Mathews, J., and Patel, D., "Gravitational Effects on Closed-Cellular-Foam Microstructure," *AIAA J. Spacecraft and Rockets*, Vol. 33, No. 2, 1996.
- [104] Bednarczyk, B. A., Aboudi, J., Arnold, S. M., and Sullivan, R. M., "Micromechanics of Spray-On Foam Insulation," Proceedings of the 43rd annual technical meeting of the society of engineering science, NASA, University Park, Pennsylvania, 2006.
- [105] Bauer, P. E. and Kummer, D. L., "Development and Performance of the Gemini Ablative Heat Shield," *AIAA J. Spacecraft*, Vol. 3, No. 10, 1966.
- [106] Milhoan, J. D. and Pham, V. T., "Thermal Certification Tests of Orbiter Thermal Protection System Tiles Coated with KSC Coating Slurries," Technical memorandum 104766, NASA Johnson Space Center, Houston, Texas, 1993, Contains a description of the Space Shuttle Orbiter insulation tiles.
- [107] Rasky, D. J., Milos, F. S., and Squire, T. H., "TPS Materials and Costs for Future Reusable Launch Vehicles," Technical report, NASA, Ames Research Center, 2000.
- [108] Sumrall, J., Lane, C., and Cusic, R., "VentureStar...Reaping the Benefits of the X-33 Program," *Acta Astronautica*, Vol. 44, No. 7-12, 1999, Discusses the design of the X-33 and its planned follow-on, VentureStar.
- [109] Gogu, C., Bapanapalli, S. K., Haftka, R. T., and Sankar, B. V., "Comparison of Materials for an Integrated Thermal Protection System for Spacecraft Reentry," *AIAA J. Spacecraft and Rockets*, Vol. 46, No. 3, 2009.
- [110] Lindell, M. C., Hughes, S. J., Dixon, M., and Willey, C. E., "Structural Analysis and Testing of the Inflatable Re-entry Vehicle Experiment (IRVE)," *Proceedings of the 47th AIAA/ASME/ASCE/AHS/ASC Structures, Dynamics and Materials Conference*, Newport, Rhode Island, 2006.
- [111] Rakow, J. A. and Waas, A. M., "Response of Actively Cooled Metal Foam Sandwich Panels Exposed to Thermal Loading," *AIAA*

- Journal* , Vol. 45, No. 2, February 2007, pp. 329, Thermal and mechanical behavior of metal foam sandwich panels for load-bearing, actively cooled structures.
- [112] Henson, G. M., “Engineering Models for Synthetic Microvascular Materials with Interphase Mass, Momentum and Energy Transfer,” *Int J. Solids and Structures*, Vol. 50, No. 14-15, 2013.
- [113] Fesmire, J. E., “Aerogel insulation systems for space launch applications,” *Cryogenics*, Vol. 46, No. 2-3, February-March 2006, pp. 111–117.
- [114] Fesmire, J. E. and Sass, J. P., “Aerogel insulation applications for liquid hydrogen launch vehicle tanks,” *Cryogenics*, Vol. 48, No. 5-6, May-June 2008, pp. 223–231.
- [115] Caogen, Y., Hongjun, L., Zhonghua, J., Xinchao, J., Yan, L., and Haigang, L., “A study on metallic thermal protection system panel for Reusable Launch Vehicle,” *Acta Astronautics*, Vol. 63, March 2008, pp. 280–284.
- [116] Johnson, M. R., “Friction Stir Welding Takes Off at Boeing,” *Welding J.*, Vol. 78, No. 2, 1999.
- [117] Cohen, H., “Space Reliability Technology: A Historical Perspective,” *IEEE Transactions on Reliability*, Vol. R-33, No. 1, 1984.
- [118] Hoffman, E., Pecquet, R., and Arbogast, W., “Compression Buckling Behavior of Large-Scale Friction Stir Welded and Riveted 2090-T83 Al-Li Alloy Skin-Stiffener Panels,” Report TM-2002-211770, NASA, 2002.
- [119] Nemeth, M., “Effects of Welding-Induced Imperfections on Behavior of Space Shuttle Superlightweight Tank,” *J. Spacecraft and Rockets*, Vol. 36, No. 6, 1999.
- [120] Davis, D. and McArthur, J., “NASA Ares I Crew Launch Vehicle Upper Stage Overview,” *Proceedings of the 44th AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit*, AIAA, AIAA, 2008.
- [121] Chang, I.-S., “Space Launch Vehicle Reliability,” *Crosslink*, October 2008.

Bibliography

- [122] Ryan, R. and Townsend, J., “Fundamentals and Issues in Launch Vehicle Design,” *J. Spacecraft and Rockets*, Vol. 34, No. 2, 1997, Discusses carbon-fiber reinforced plastic (composite) tanks for liquid oxygen and hydrogen storage in launch vehicles.
- [123] Harris, C. E., Stuart, M. J., and Gray, H. R., “A Survey of Emerging Materials for Revolutionary Aerospace Vehicle Structures and Propulsion Systems,” Report TM-2002-211664, NASA, Hampton, Virginia, 2002.
- [124] Bar-Cohen, Y., editor, *Biomimetics: Biologically Inspired Technologies*, CRC Taylor & Francis, Boca Raton, Florida, 2006.